University of Virginia

Two Designs for an Orbital Transfer Vehicle

USRA/NASA Advanced Design Program
Summer Conference June 1988

Report 3 of 3

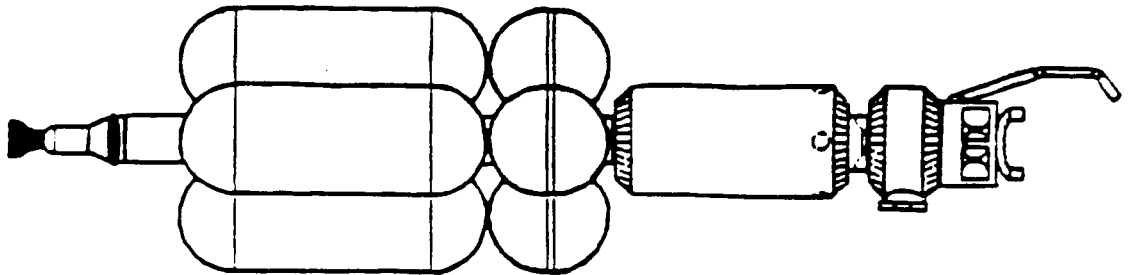Conceptual Design of a Manned
Orbital Transfer Vehicle

prepared by the
MOVERS
for AE441-2 Space Vehicle Design

and

NASA/USRA Advanced Design Program

at the
University of Virginia
May 1988

MODULAR ORBITAL VEHICLE ENGINEERING RESEARCH SOCIETY

RICHARD DAVIS   Group Leader / Propulsion

MILES DUQUETTE   Design / On-Board Computer Systems

REBECCA FREDRICK   Orbital Mechanics / Navigation Systems

DANIEL SCHUMACHER   Human Factors / Interior Design

SCHAEFFER SOMERS   Servicing System / EVA

STANLEY STAFIRA   ECLSS / Power Systems

JAMES WILLIAMS   Reaction Control / Tankage

MARK ZELINKA   Shielding / Structures

TABLE OF CONTENTS

I. OVERVIEW

With the advent of the manned space station, man now requires a spacecraft based on the space station with the ability to deploy, recover, and repair satellites quickly and economically. Such a craft would prolong and enhance the life and performance of many satellites. We at MOVERS feel that the time for such a spacecraft has arrived, and have developed a basic design for an orbital transfer vehicle, or OTV.

The basic design criteria determined for the OTV are as follows: The craft must be able to deliver and retrieve from geosynchronous orbit (GEO) a payload of 15,000 pounds. It must also be able to sustain a crew of three for seven days, and support extra-vehicular activities (EVA). The basic spacecraft should be adaptable to earth-moon missions with payloads as large as 80,000 pounds.

Results indicate that our OTV, which satisfies the above criteria, will be modular in design. For the basic mission, the low-earth orbit (LEO) to GEO transfer, the OTV consists of a command module, a habitability module, an airlock, a remote manipulator system (RMS), and EVA flight support station, propellant tanks, and an engine. A schematic of our OTV is displayed by Figure 1. To achieve the longer lunar missions, additional modules and tanks are easily attached.

MOVERS researched design of the OTV and her systems in the following areas: avionics, crew systems, electrical power systems, environmental control/life support systems, navigation and orbital maneuvers, propulsion systems, reaction control systems (RCS), servicing systems, and structures. The basic

ideas contained in each section are summarized below.

The avionics section discusses the state-of-the-art equipment, both hardware and software, which was chosen for the OTV. New features of the computer system include bubble memory and electroluminescent screens, while all of the software will use Ada programming language.

The crew systems report examines the relationship between man and machine in space, with emphasis on the need for human factors research and application of that research to the OTV. The report gives consideration to habitability, psychology and behavioral science, and design of the OTV in order to optimize crew satisfaction, work efficiency, and the success of future space missions.

A chemical power production system will provide the power for the OTV. It uses two Hydrogen-Oxygen fuel cells to produce the electrical power needed by the spacecraft. The environmental control/life support system will be integrated with the OTV's power production system. The craft will operate with a partially closed system. The system receives water from the fuel cell operation, and regenerate the CO_2 produced in the environment into elements that can be used again in the OTV's atmosphere.

For navigation the OTV will employ a combination of reliable instruments from the space shuttle and recently developed state-of-the art equipment. The navigation section also details the required orbital maneuvers for a typical mission, including a solution for the difficult rendezvous maneuver, termed the HITME

maneuver.

We at MOVERS elected to employ nuclear power on our OTV as the wave of the future. Our propulsion specialist chose a high thrust, NERVA (Nuclear Engine for Rocket Vehicle Application) derivative engine. The engine, including the neutron/gamma shield, weighs 12,500 pounds, has a specific impulse of 880 seconds, and can deliver 30,000 pounds thrust. For the LEO to GEO mission, utilizing high thrust nuclear engines results in significant propellant savings over traditional chemical systems. These engines were also found to be very competitive with proposed, aerobraked, chemical systems. Environmental analysis indicated that the problems of catastrophic failures and the diffusion of radioactive particles through the fuel rods in LEO does not pose significant health hazards to the population of the earth. The issue of misfired burns does represent a possible health hazard, and this problem is addressed in the propulsion systems section.

The RCS report studied three RCS systems for the OTV to satisfy the six degree of freedom requirement. These RCS' were (1) monomethylhydrazine/nitrogen tetroxide, (2) supercritical hydrogen/oxygen, or cryogenics, and (3) monopropellant hydrazine. The cryogenic RCS was selected due to its higher performance and lower total program cost. The optimum configuration of the propellant tanks for the nuclear propulsion, non-aerobraked OTV were three cylindrical tanks; the cylinder shape was chosen because it gave the best propellant quantity for packaging within the shuttle cargo bay.

4

The servicing systems section examines the operations and hardware elements required to fulfill the spacecraft servicing function of the OTV in geosynchronous orbit. The report presents a design that is capable of resupplying fluid consumables to orbiting spacecraft and replacing malfunctioned or obsolete components. The design utilizes advanced telerobotic technology to execute the servicing operations with manual Extra-vehicular activity (EVA) as a backup. The servicing system is modular and can be separated from the core OTV vehicle.

The structures section concentrates mainly on shielding, mostly radiation shielding with some meteorite shielding. The four major sources of radiation are 1) cosmic flares, the rarest but most damaging type, 2) the Van Allen Belts, the worst as far as dose/time goes and the second worst in terms of total dose, 3) the trapped radiation from the Starfish thermonuclear explosion of 1962, similar to the Van Allen Belts, and 4) background cosmic radiation, which is negligible. The total dose expected for a one week mission is around 35 rems for a $5g/cm^2$ aluminum shield. In case of a solar flare during flight, the spaceship will turn the reactor shield towards the sun to protect the OTV.

A typical mission of the OTV might occur as follows (refer to the navigation section for definitions and figures detailing the maneuvers):

April 24, 1996

The OTV and crew is called upon to service a failing Telstar satellite. After preparing the OTV for departure, the crew waits for Telstar to cross into the Initial Launch Zone (ILZ). The dry mass of the OTV is 50,300 pounds, and 84,926 pounds of propellant were added for estimated usage. The total mass of the OTV upon departure is 135,226 pounds, utilizing three cylindrical tanks for propellant storage.

12:00 PM - The Telstar is in the ILZ and the OTV crosses the line of nodes at the initial departure point. Telemetry gives the initial angle of Telstar as 33°. The parameters for the waiting ellipse are quickly calculated to be:

Semi-major axis, A - 27,628,030 ft

Eccentricity, e - 0.21

Initial required /\V - 2543 ft/s

The time-of-flight is 2 hours and 11 minutes.

2:11 PM - The OTV once again crosses the initial departure point. The burn is then performed to place the OTV into a Hohmann transfer for rendezvous. This /\V is again provided by the computer as 5439 ft/s. The time-of-flight of the Hohmann ellipse is 5 hours and 16 minutes.

7:27 PM - The OTV soft docks with Telstar, after performing a final /\V of 6018 ft/s. Servicing begins, using the servicing

6

system platform. The servicing takes 4.5 days.

April 29, 1996

Servicing is completed. After a night's sleep, the OTV and crew prepare to return.

7:30 AM - The OTV crosses the line of nodes opposite of the point of rendezvous with Telstar. The initial $/\backslash V$ is 6018 ft/s, placing the OTV into the return Hohmann transfer ellipse. Again, time-of-flight is 5 hours and 16 minutes.

12:46 PM - The OTV reaches LEO on the line-of-nodes at the final rendezvous point. Telemetry gives the angle of the space station to this point as 51°. The computer provides the parameters for the waiting ellipse:

A - 23,447,540 ft

e - 0.069

Required $/\backslash V$ - 7116 ft/s

The time-of-flight of the waiting ellipse is 1 hour, 43 minutes.

2:29 PM - The OTV meets the space station, mission complete. The total elapsed mission time was 5 days, 2 hours and 29 minutes.
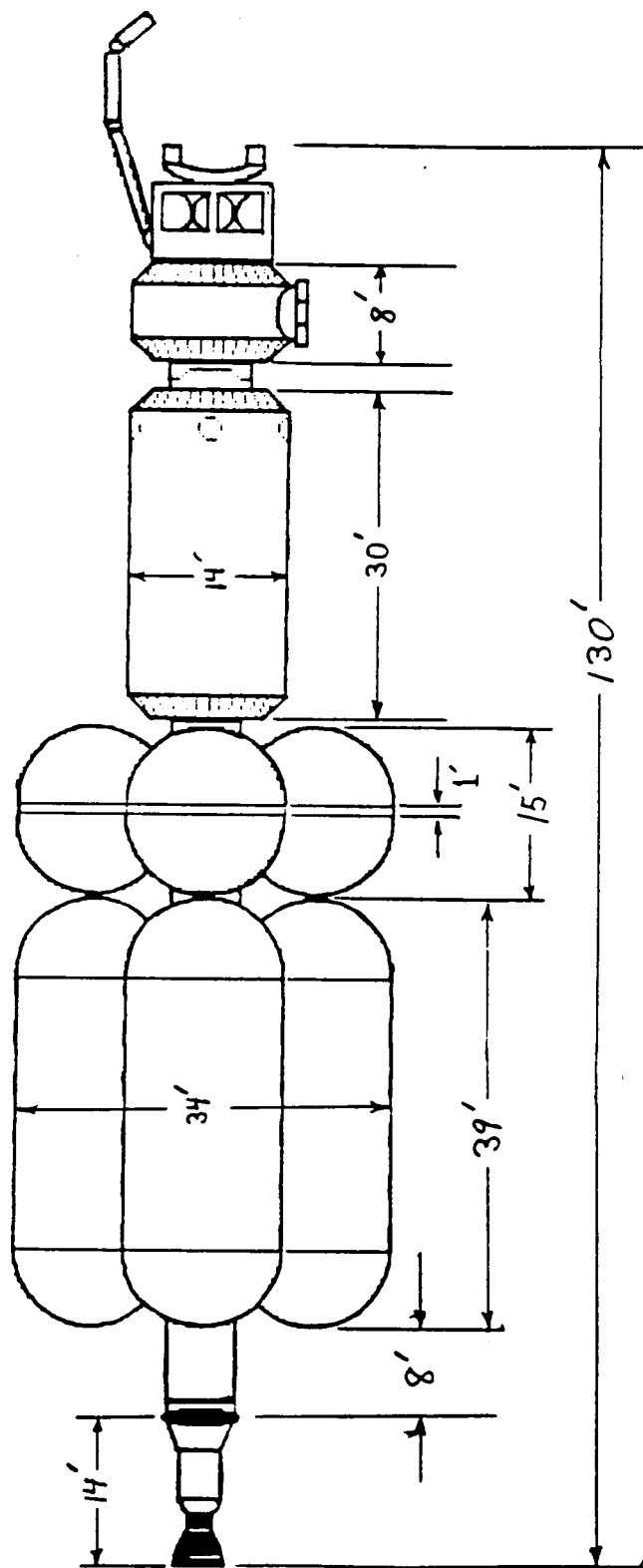
Design Integration

Appearing on the next page is a diagram of the OTV configuration proposed by this report. Looking from right to left, this includes the satellite servicing system, the command module, the living quarters module, the four secondary propellant tanks, the four main propellant tanks, the reactor shield, the nuclear engine, and the exhaust nozzle. Each of these elements are discussed in detail throughout the rest of this study.

The addition of a 15,000 lb payload results in the need for the secondary propellant tanks (see the section on Tankage) to be filled with propellant.

The table below contains some overall dimensions of the OTV configurations.

Length (overall)	130 feet
Width (maximum)	34 feet
Command Module (length/dia.)	8 / 14 feet
Living Quarters (length/dia.)	30 / 14 feet
Main Propellant Tanks (length/dia.)	4 tanks: 39/14 feet
Secondary Propellant Tanks (length/dia.)	4 tanks: 15/14 feet
Engine (length/dia.)	14 / 4 feet

OTV Design Configuration

$1\,cm \approx 6.86\,ft$

II. AVIONICS SYSTEM

Introduction

The term "avionics" refers to all the specialized electrically powered and operated equipment used by the OTV to perform tasks ranging from orbital tracking and navigation to communication and data management. Due to the nature of the OTV application, the structure of the avionics system has the following general areas:

- Data Management
  - Computer Processing (incl. program implement.)
  - Computer Interfacing and Control of the other
                                    avionics components
  - Data Acquisition
- Navigation and Control Systems
- Communication and Tracking Systems
- other miscellaneous electrical equipment

Some of these systems operate independently of the others, leaving the power supply as the only element common to all of them. However, most of these areas do tie in directly with each other, as can be seen on the figure below.

Navigation and Control Systems (NCS)	Data Management Systems (DMS)	miscellaneous electrical equipment

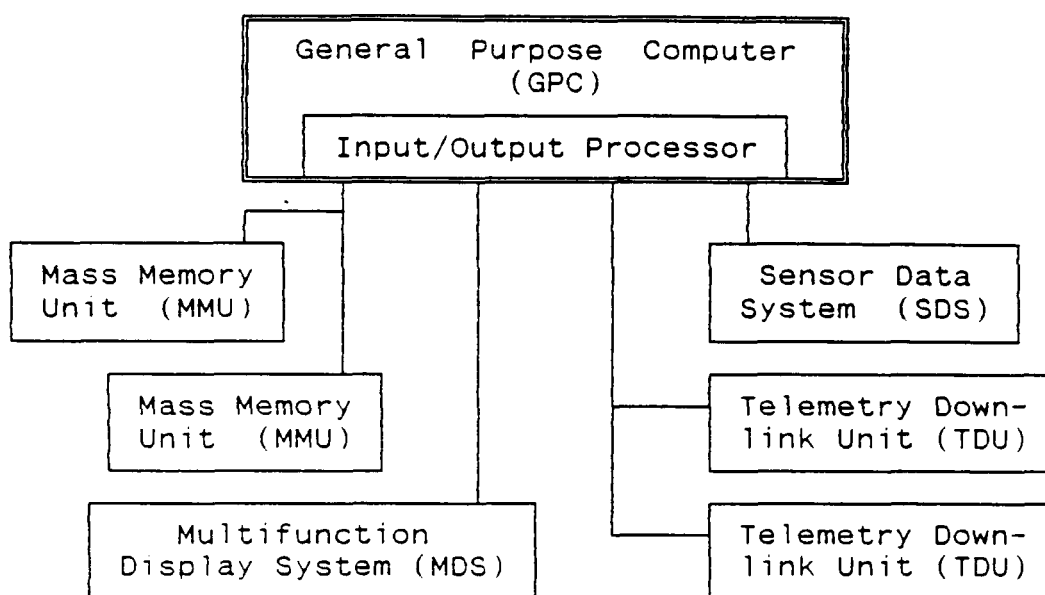| Communications and Tracking Systems (CTS)	other independ. miscellaneous equipment
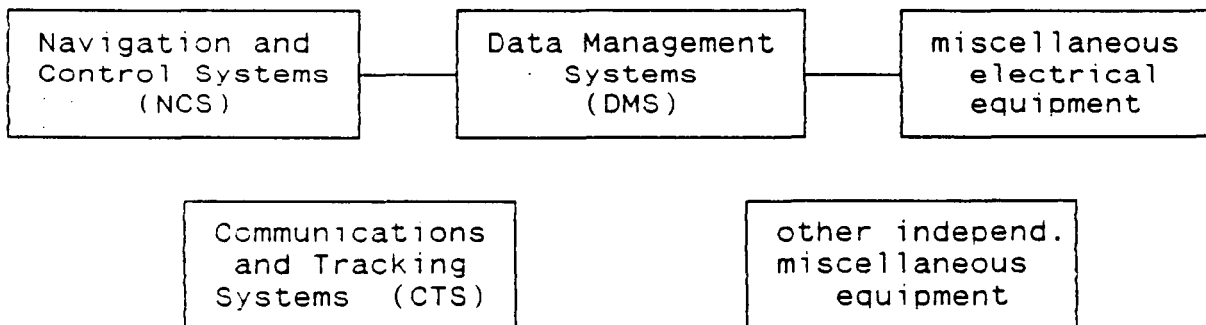
OVERALL VIEW OF AVIONICS SYSTEMS

Each of these systems will now be explained in further detail.

11

1. DATA MANAGEMENT SYSTEM (DMS)

The DMS is comprised of many individual components tied directly to a General Purpose Computer (GPC), the "brains" as it were, of the DMS. The particular components of the DMS are shown below.

```
        ┌─────────────────────────────────────┐
        │   General   Purpose   Computer       │
        │            (GPC)                     │
        │  ┌───────────────────────────────┐   │
        │  │   Input/Output Processor      │   │
        │  └───────────────────────────────┘   │
        └─────────────────────────────────────┘

  ┌──────────────────┐              ┌──────────────────┐
  │  Mass Memory     │              │  Sensor Data     │
  │  Unit  (MMU)     │              │  System  (SDS)   │
  └──────────────────┘              └──────────────────┘

      ┌──────────────────┐          ┌──────────────────┐
      │  Mass Memory     │          │  Telemetry Down-  │
      │  Unit  (MMU)     │          │  link Unit (TDU)  │
      └──────────────────┘          └──────────────────┘

    ┌──────────────────────┐        ┌──────────────────┐
    │  Multifunction       │        │  Telemetry Down-  │
    │  Display System (MDS)│        │  link Unit (TDU)  │
    └──────────────────────┘        └──────────────────┘
```

DATA MANAGEMENT COMPONENTS

The GPC contains a central processing unit (CPU) and an input/output processor, just as most computers do. These parts of the GPC control the flow and processing of the data acquired from the other avionics systems. The mass memory units (MMU's) are radiation hardened, nonvolatile storage cells where all the maintenance/application programs, data look-up tables, and collected sensor data are stored. There are two of these units

for purposes of redundancy. The telemetry downlink units (TDU's) are encoder/decoder and transmitter/receiver systems which create and control the flow of computer collected data to and from either the space station or the Earth. Again, there are two of these units for redundancy.

The multifunction display system (MDS) is the "human" interface of the DMS. The components of the MDS include display screens, keyboard units, and display controller units which control the screen output and interpret the keyboard input. There are two complete MDS systems for multiple accessing and redundancy.
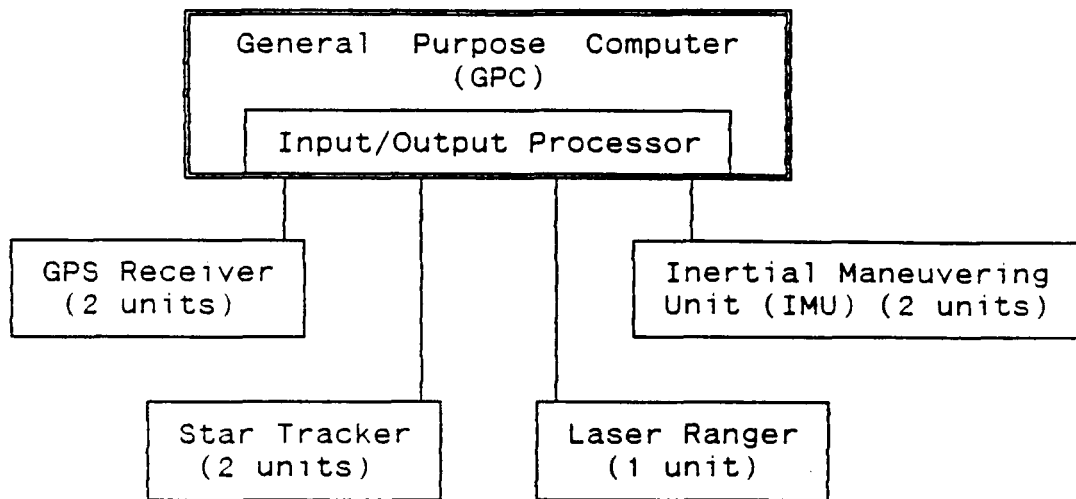
Also shown on the next page is a schematic of the sensor data system (SDS). The purpose of the SDS is to collect sensor impulses (usually electrical) of several element properties at various points around the OTV and then convert those impulses to data forms that the DMS can understand. The three properties measured by the SDS are temperature, pressure, and strain. In the case of the OTV there will be approximately:

<div style="text-align: center;">

	50	thermocouples
	30	pressure sensors
and	50	strain gauges.

</div>

The thermocouples will produce very small voltages which will be converted to larger corresponding voltages by the thermocouple processing unit (TCU). The pressure and strain gauge information will be interfaced by two processing control units (PCU's), one 64 channel and one 16 channel PCU.

13

2. NAVIGATION AND CONTROL SYSTEM (NCS)

The purpose of the NCS is to provide navigational guidance and attitude control of the OTV. The NCS is linked directly to the GPC as shown below. For more information on the NCS, see the section on navigational systems.

```
          ┌─────────────────────────────────────┐
          │  General   Purpose   Computer        │
          │            (GPC)                     │
          │  ┌───────────────────────────────┐   │
          │  │    Input/Output Processor     │   │
          │  └───────────────────────────────┘   │
          └──────┬────────┬────────┬────────┬────┘
                 │        │        │        │
    ┌────────────┴──┐     │        │     ┌──┴──────────────────────┐
    │ GPS Receiver  │     │        │     │ Inertial Maneuvering    │
    │ (2 units)     │     │        │     │ Unit (IMU) (2 units)    │
    └───────────────┘     │        │     └─────────────────────────┘
              ┌───────────┴──┐  ┌──┴───────────┐
              │ Star Tracker │  │ Laser Ranger │
              │ (2 units)    │  │ (1 unit)     │
              └──────────────┘  └──────────────┘
```

NAVIGATION AND CONTROL SYSTEM

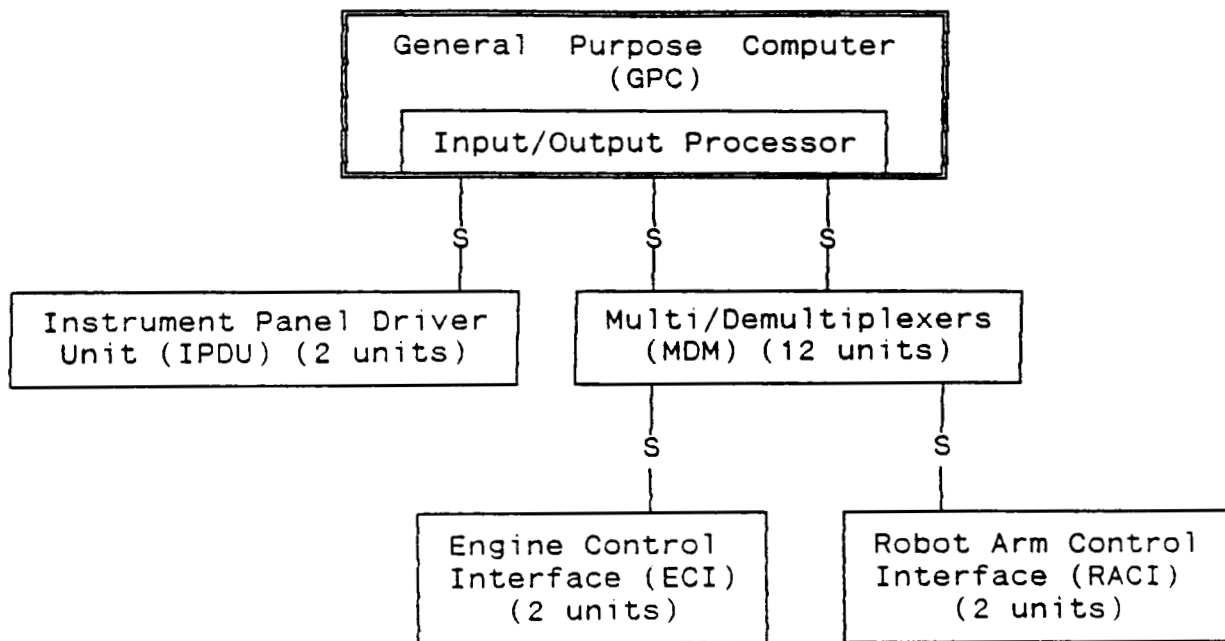## 3. COMMUNICATION AND TRACKING SYSTEM (CTS)

The elements of the CTS are radio's, televisions', antennas, etc., which are used for OTV tracking and crew - space station communication. This specific elements of the CTS are:

> S-Band PM (radio)
> Television (for docking)
> UHF
> Antennas
> Support Equipment

This system and its elements run independently of the DMS.

14

4. MISCELLANEOUS ELECTRICAL EQUIPMENT

There are two types of components in this area: those that are and are not DMS interfaced. Those interface units which are connected to the DMS are shown below. The "S" associated with the data bus lines refers to "string" data bus, i.e., these lines

```
        +-------------------------------------+
        |  General   Purpose   Computer       |
        |           (GPC)                     |
        |  +-------------------------------+  |
        |  |    Input/Output Processor     |  |
        |  +-------------------------------+  |
        +-------------------------------------+
              |           |           |
              S           S           S
              |           |           |
   +--------------------+ +---------------------------+
   | Instrument Panel   | | Multi/Demultiplexers      |
   | Driver Unit (IPDU) | | (MDM) (12 units)          |
   | (2 units)          | |                           |
   +--------------------+ +---------------------------+
                              |              |
                              S              S
                              |              |
                   +------------------+ +------------------+
                   | Engine Control   | | Robot Arm Control|
                   | Interface (ECI)  | | Interface (RACI)  |
                   | (2 units)        | | (2 units)        |
                   +------------------+ +------------------+
```

MISCELLANEOUS DMS INTERFACES

are triple redundant data buses. As with all vehicles the OTV has an appropriate number of instrument panel displays, switches and controls for operation of the ship. These panel controls and displays are driven and controlled by the DMS through the instrument panel driver unit (IPDU), of which there are two for purposes of redundancy. Another component shown above is the engine control interface (ECI). This unit is extremely important for precise operation of the propulsion and attitude control

15

systems, and so there are two complete ECI units which the DMS can use to control the engines. The third interface connected to the DMS is the robot arm control interface (RACI). The robot arm will be capable of both manual and automatic control. For the latter type of control, the DMS will access the mechanical systems of the robot arm by means of the RACI. Again, there are two redundant RACI units. Because these last two interfaces are triple-stringed, and therefore require three times the normal amount of data bus cabling, and because they are not in close proximity to the DMS, there appears in the figure above twelve units called multi/demultiplexers (MDM). These components convert data bus signals to and from serial and parallel formats. This enables the use of less cabling between the interfaces and the DMS, and thus less weight. There are twelve of these units because there are six total data bus lines (three for each interface), and each line needs one MDM at each end.

Some of the other miscellaneous electrical equipment which does not connect with the DMS are such things as the robot arm itself, the instrument panels themselves, various lights, and so forth.

Table 1.

UPGRADED SYSTEM COMPONENT SUMMARY

COMPONENT TYPE				POWER (watts)		WEIGHT (lbs)		VOLUME (cu. in.)	
				per unit	total	per unit	total	per unit	total
DATA MANAGEMENT SYSTEM									
General Purpose Computer								1210	3600
Mass Memory Unit								600	1200
									190
							120	100	200
				500	500			2000	4000
						5.5	5.5	144	144
						16	16	448	448
						4.3	4.3	150	150
							40	300	400
							36	572	1144
							30	6100	6100
								2100	2100
							36	1100	1100
								1000	2000
							100	800	1600
							10	800	1600
							6	45	780
						0.3	25	700	700
SUBTOTALS:				835	1346		464	27546	27546
COMMUNICATIONS AND TRACKING SYSTEM									
S-Band PM				700	700		129	8250	8250
UHF				250	20		48	1350	1350
Support Equipment				200	200		82	2705	2705
Antennas							106		500
Cables and Mountings							50	500	500
SUBTOTALS:				925	925		515	13801	13801
POWER / DISTRIBUTION									
Instrument Power, Controls, batteries, etc.				50	50		60	26000	26000
GRAND TOTALS:				1810	2521		1039	46347	46347

This section outlines the power, weight, and volume requirements for each of the avionics components and the reasons for the selection of particular systems over that of others. A summary of all the avionics components and their specifications is shown in Table 1 on the next page. It is important to note that this figure includes all of the aforementioned' components with the exception of the robot arm. The sources of this information, as well as all the data described thus far, appear in the reference list at the end of this section. Several of the systems shown in the figure deserve some explanation about their specifications. These are described below. The preeminent requirement of all these components is reliability, followed closely by the minimization of power, weight, and volume.

1. GENERAL PURPOSE COMPUTER

The GPC chosen for the OTV mission is the Control Data (SCP). This computer was chosen over others for the following reasons. To begin with, the nature of the OTV mission requires that the data processing system be the most up-to-date as possible. This is due to not only the need for the most reliable yet lightest, smallest, least power consuming, and fastest computer, but also the need for compatibility with future trends. Such trends will include the adoption of the Air Force 1750A Standard ISA and the increased use of CMOS/SOS (Complementary Metal Oxide Semiconductor/Silicon-On-Sapphire) chip technology

[2], [7]. This trend is can be seen on the soon-to-be-launched Galileo project which uses CMOS chip technology. The eight candidate computers for the OTV mission are shown in Table 2 on the next page, along which each one's specifications. Analysis of the characteristics of each computer led to the final choice of the Control Data (SCP).

2. MASS MEMORY UNIT

The type of mass memory technology selected for the OTV mission is Bubble Memory. This was chosen after analyzing the available memory technologies, as shown in Table 3. The most important features of this figure are reliability and radiation hardness. In short, the bubble memory technology was selected for reasons such as those stated by Greenberg, et.al., "the advantages of solid-state bubble memory over traditional nonvolatile tape recorders are semi-random access of data, millisecond access time, high reliability, radiation hardness, and low power." [6,p.35]

3. MULTIFUNCTION DISPLAY SYSTEM

The type of display selected for the OTV mission is electroluminescence flat-panel technology (EL, for short). This is a break from the traditional CRT's which the space shuttle uses. The justification for this is based upon the great technological advances that have been made in this area in recent years. Table 4 summarizes the comparisons made between CRT's, EL's, Liquid Crystal, and Plasma Panel displays. As shown on the

EVALUATION OF FLAT PANEL TECHNOLOGIES *

AREA	VALUE (see below)	Cathode Ray Tube	Electro-lumi-nescent	Liquid Crystal	Plasma Panel
SHAPE AND OPERATING CHARACTERISTICS					
Power Consumption	H	Fair	Fair-good	Excellent	Fair
Screen Size	M	Excellent	Fair	Fair	Fair
Depth	M	Poor	Excellent	Excellent	Good
Weight	H	Poor	Excellent	Excellent	Excellent
Ruggedness	M	Fair-good	Good-excel	Excellent	Excellent
Temperature Range	H	Good	Excellent	Fair-good	Good
IMAGE QUALITY					
Brightness	L-M	Excellent	Excellent	Fair-good	Good
Resolution	L-M	Good-excel	Good	Fair-good	Good
Contrast	L-M	Good-excel	Excellent	Fair	Good
Gray Scale	L	Excellent	Fair	Poor	Poor
Viewing Angle	L	Excellent	Good	Poor	Good
Color Capability	L	Excellent	Poor	Poor	Poor
Image Stability	M	Fair	Excellent	Good-excel	Excellent
COST	L-M	Low	Med-High	Low	Med-High

VALUE: Level of Importance: L = Low,   for OTV application.
                             M = Moderate,
                             H = High,

* Source: Buxton & Baecker: SIGGRAPH '86,
           and Tannas,L.E.,"Electroluminescence Cathes the Public Eye".

Table 4.

figure, each display was given a rating in each of the areas listed. In addition, the importance of each area with respect to the nature of the OTV mission is shown (see "VALUE"). As can be seen on the figure, although EL's do not fair well in terms of image quality, they have excellent ratings for shape and operating characteristics, which is the most important for the OTV. The actual specifications of the EL displays shown on Table 1 were obtained from Table II of reference [15].

4. DATA BUS NETWORK

The data bus network is the system of cabling used to interconnect all the components of the DMS together. This system uses both single and triple string data busing with all external components connected to each of the three GPC's. A diagram of the data bus network for the OTV is shown following Table 4.

Table 2.

SPACECRAFT MEMORY TECHNOLOGY SURVEY

Source: Goodenough, et al, "Survey of Spacecraft Memory Technologies"

	RAMs	ROMs/PROMs	EEPROM	MAGNETIC BUBBLE MEMORY	MAGNETIC CORE	PLATED WIRE MEMORY	MAGNETIC TAPE RECORDER
Storage element	4K, 16K, 64K bits/chip	8K, 16K, 64K bits/chip	16K, 32K, 64K bits/chip	1M, 4M bits/chip	Cores 9-13 mil ferrite 64x16 bits	PWM 2mil 64Kx16 Bits	10^9 Bits (.25 to .001 in)
Access time	100/400 nsec	30/85, 300/400	200/400 nsec (read only)	40-50 msec (average access)	350/900 msec	600/900 nsec	msec sequential access
Programmability	Easy, fast	Bit oriented bipolar	10^4 erase/write slow	Slow, semi-random	Easy, fast	Easy, fast	Easy, slow
Operating temperature range	-55 C to +125 C	-55 C to +125 C	-55 C to +125 C	-55 C to +85 C	-55 C to +85 C	-40 C to +85 C	0 C to +50 C
Power requirement and dissipation	+5V, 200mW bipolar	+5V, 200mW bipolar	+5V, some 12/24V	+5V, +15V, +200, (20W)	+5V, +12V, (70W, 13mil, 50W, 9mil)	+5V, -12V (80W)	200 - 300 (4W read, 8W write)
Volatility	Volatile	Nonvolatile	Nonvolatile	Nonvolatile	Nonvolatile	Nonvolatile	Nonvolatile
Level of radiation hardness	10^4 to 10^6 Rads	10^4 to 10^6 Rads	10^4-10^6 Rads	10^4-10^6 Rads	10^4-10^6 Rads	10^4-10^6 Rads	10^4-10^6 Rads
Reliability	High	High	High	High	High	High	Medium

Table 3.

Software

1. REQUIREMENTS

The nature of the OTV mission will place demands on the DMS similar to those of communication satellites. A typical software system for these satellites, and the OTV, has the following requirements: [5,p.43]

1) Orbit Prediction - for planning launch windows and orbital maneuvers

2) Orbit Determination - for accurate determination of current orbital elements

3) Apogee Maneuver Planning - for transfer orbit planning

4) Stationkeeping - to make orbit corrections due to perturbative interferences

5) Attitude Determination - spacecraft orientation

6) Attitude Maneuver Planning - for planning attitude adjustments

7) Maneuver Commanding - to determine proper thrust times and propellant required

8) Database Management - to organize all the software and data acquisition operations

2. LANGUAGE

In the past, assembly language coding of software has been the most widely used, due to the limited memory capacity of the on-board computers and the lack of off-the-shelf compilers. This changed somewhat with the Shuttle, which uses a higher order language (HOL) called HAL/S, which was developed by Intermetrics Inc. between 1970 and 1972. HAL/S, however, is losing the

popularity battle to Ada, another HOL that is 10 years newer than HAL/S and which will soon be adopted by DoD as their standard spacecraft software (MIL-STD-1815A). HAL/S and Ada were derived from two very different concepts. Whereas HAL/S was designed for the special purpose of flight software, Ada was developed to treat a wider range of applications. In addition, Ada has a decade of advances in programming technologies upon which to build. In his article Space Station Flight Software: HAL/S or Ada?, Allan Klumpp "recommends that Ada be considered for the primary programming language and that HAL/S be retained in order to utilize core software inherited from the shuttle." [8,p.20] Therefore, the OTV will use Ada as its primary software language.

3. ARTIFICIAL INTELLIGENCE AND EXPERT SYSTEMS

The latest development in the area of software for space applications has arisen out of NASA's drive to develop a permanent space station. The research being conducted to develop operational artificial intelligence (AI) systems for the space station has led to the advancement of expert systems which simulate a human expert of a particular subject. Expert systems under development at Johnson Space Center include: [10,p.59]

-- On-orbit position determination for the space shuttle and space station.

-- Control of a spacecraft electrical system.

-- Diagnosis of software failures by ground control during a mission.

-- Planning for a space shuttle or other spacecraft mission, including trajectories and attitudes.

      -- Processing of radar tracking data during shuttle
          ascent and reentry.

      -- Control of a system to remove carbon dioxide from
          spacecraft cabin air and control cabin pressure.

In addition, LinCom of Nassau Bay, Texas is developing expert systems for automated docking procedures of robotic spacecraft. These efforts have been encouraged through a report in 1985 by the National Aeronautics and Space Administration Advanced Technology Advisory Committee called "Advancing Automation and Robotics for the National Space Program". This report urged the continual development of AI software and robotics systems for the space station, in such areas as: [11,p.63]

      -- Electrical power expert systems that can distribute
          loads, orient solar arrays, provide analysis of
          electrical system performance trends and provide
          fault diagnosis.

      -- Communication and tracking systems, including com-
          munication scheduling, rendezvous tracking and data
          rate selection.

      -- Information and data management expert systems for
          control of subsystem status, redundancy and config-
          uration management and data-base management.

      -- Environmental control and life support systems, in-
          cluding crew alarm, station atmosphere monitoring
          and control, and hyperbaric chamber control.

Some of these tasks will be included in the functions of the DMS for the OTV, however, due to the limited microprocessing space of the OTV, most of these AI and ES capabilities will have to be omitted.

REFERENCES

1. Bragg, B. J., et al., <u>A Design Study for an Aeroassist Flight Experiment</u>, NASA-Johnson Space Flight Center, June 1985, Appendix D.

2. Byington L. and Theis D., "Data-Processing Hardware for Spacecraft - Air Force Standard 1750A-ISA is the New Trend," <u>Computer</u>, Vol 19, No.11, Nov 1986, pp 50-59.

3. Carney, P., "Selecting On-Board Satellite Computer Systems," <u>Computer</u>, Vol 16, No.4, Apr 1983, pp 35-41.

4. Dawson, W. K. and Dobinson, R. W., "A Framework for Computer Design," <u>IEEE Spectrum</u>, Vol 23, Oct 1986, pp 49-54.

5. Ekman, D., "Orbit Control Software for Communications Satellites," <u>Computer</u>, Vol 16, No.4, Apr 1983, pp 43-51.

6. Greenberg, E., et al., "Survey of Spacecraft Memory Technologies," <u>Computer</u>, Vol 18, No.3, Mar 1985, pp 29-39.

7. Iversen, W.R., "Control-Data Launches CMOS-SOS Space Computer," <u>Electronics</u>, Vol 59, No. 26, July 1986, pp 27-28.

8. Klumpp, A., "Space Station Flight Software: HAL/S or Ada?," <u>Computer</u>, Vol 18, No.3, March 1985, pp 20-28.

9. <u>Manned Geo Sortie Study</u>, NASA-Johnson Space Flight Center, 1983

10. Marsh, A. K., "NASA Space Station Effort Drives Expert Systems Research at Johnson," <u>Aviation Week and Space Technology</u>, Vol 122, No.15, April 22, 1985, pp 59-63.

11. "NASA Report Urges Developing Robotics, Software for Station," <u>Aviation Week and Space Technology</u>, Vol 122, No.15, April 22, 1985, page 63.

12. Rennels, D., "Reconfigurable Modular Computer Networks for Spacecraft On-Board Processing," <u>Computer</u>, Vol 11, No.7, July 1978, pp 49-59.

27

13. Ross, Cindy, _State-of-the-Art Report: Data Management Systems_, Dept. Mech. & Aero. Engineering, Univ. of Va., December 12, 1986.

14. Suffredini, M. T., _Data Processing System Overview Workbook_, NASA-Johnson Space Flight Center, DPS OV 2102, January 1984.

15. Tannas, L.E., Jr., "Electroluminescence Catches the Public Eye," _IEEE Spectrum_, Vol 23, Oct 1986, pp 37-42.

16. Theis, D., "Spacecraft Computers: State-of-the-Art Survey," _Computer_, Vol 16, No.4, Apr 1983, pp 85-97.

III. HUMAN FACTORS

Human factors is a combination of all equipment required to maintain the crew in a healthy, comfortable, productive working environment. According to the RFP, the OTV requires systems to support three men for a period of seven days. The majority of the equipment used will be very similar to the equipment employed by the Space Shuttle and equipment tentatively planned for use on the Space Station.

Psychological Considerations

When designing the spacecraft the following considerations must be taken into account: privacy, crowding, and sensory deprivation. Crewmembers should be allowed to select a variety of clothes and colors for their wardrobe. This would brake up the monotony if everyone wore the same clothing. During Skylab missions, there was a lack of fragrances - only lemon dishwashing detergent and a spice scented deodorant were present (1:31). More items should be included that will provide a greater variety in aromas present. Additional items might include a variety of deodorants, aftershaves, and detergents. Also, the OTV will be broken up into two separate compartments, enabling a crewmember to separate himself from the other crewmembers in a time when privacy is desired. Estimates predict that each crewmember will need 180 cubic feet for total habitable volume, and 41 cubic feet for private crew quarters (5:10).

CMMD/CTRL WORK-STATION HMF EXERCISE STATIONS STOWAGE CREW QUARTERS HMF MEDICAL LAUNDRY PMAD FMAD GALLEY ECLSS

FORE

PMAD FMAD WASTE MGMT PERSONAL HYGENE WARDROOM CREW QUARTERS ECLSS - THC ECLSS - AR HYGENE CONTROLS POTABLE ECLSS - H20

STARBOARD SIDE	FLOOR	PORT SIDE	CEILING
PMAD FMAD	PMAD FMAD	Waste Management	ECLSS
Laundry/ Housekeeping	ECLSS - THC	Personal Hygene	HMF Excercise
Galley	ECLSS - AR	Cmmd/Ctrl Workstation	Stations
Galley	ECLSS - Hygene H2O		Stowage
Galley	ECLSS - H2O & Controls	Wardroom	Stowage
Galley	ECLSS - Potable H2O		HMF - Medical
Safe Haven	Crew Quarters	Crew Quarters	Crew Quarters

Figure 3-1.
Interior Layout

Interior Design

The interior design is set up in a one-gravity configuration. It consists of 21 service modules each 42 inches wide. The subsystem equipment is installed in modular packages that occupy the space near the walls, leaving a 84 inch square opening in the center to be occupied by the crew. The inboard faces of the crew quarters form a square "hallway" about 60 inches in width through which personnel and material traffic move without disturbing the sleeping crewmembers.

Crew Quarters

A very important requirement for the mission is the capability of each crewperson to have a private retreat that belongs to him/her alone, as well as one that provides noise and light control for restful sleeping and relaxation (see Figure 3-2). Each crew compartment displaces one and a half service bays and encloses approximately 150 ft^3. The zero-gravity environment is exploited to make effective use of a relatively limited volume by keeping the sleep restraint and personal use console oriented parallel. A small window is available for personal recreational viewing. A personal use console is installed, providing storage containers, a video/audio/data processing center for private work or entertainment, and appropriate body restraints (2:173).
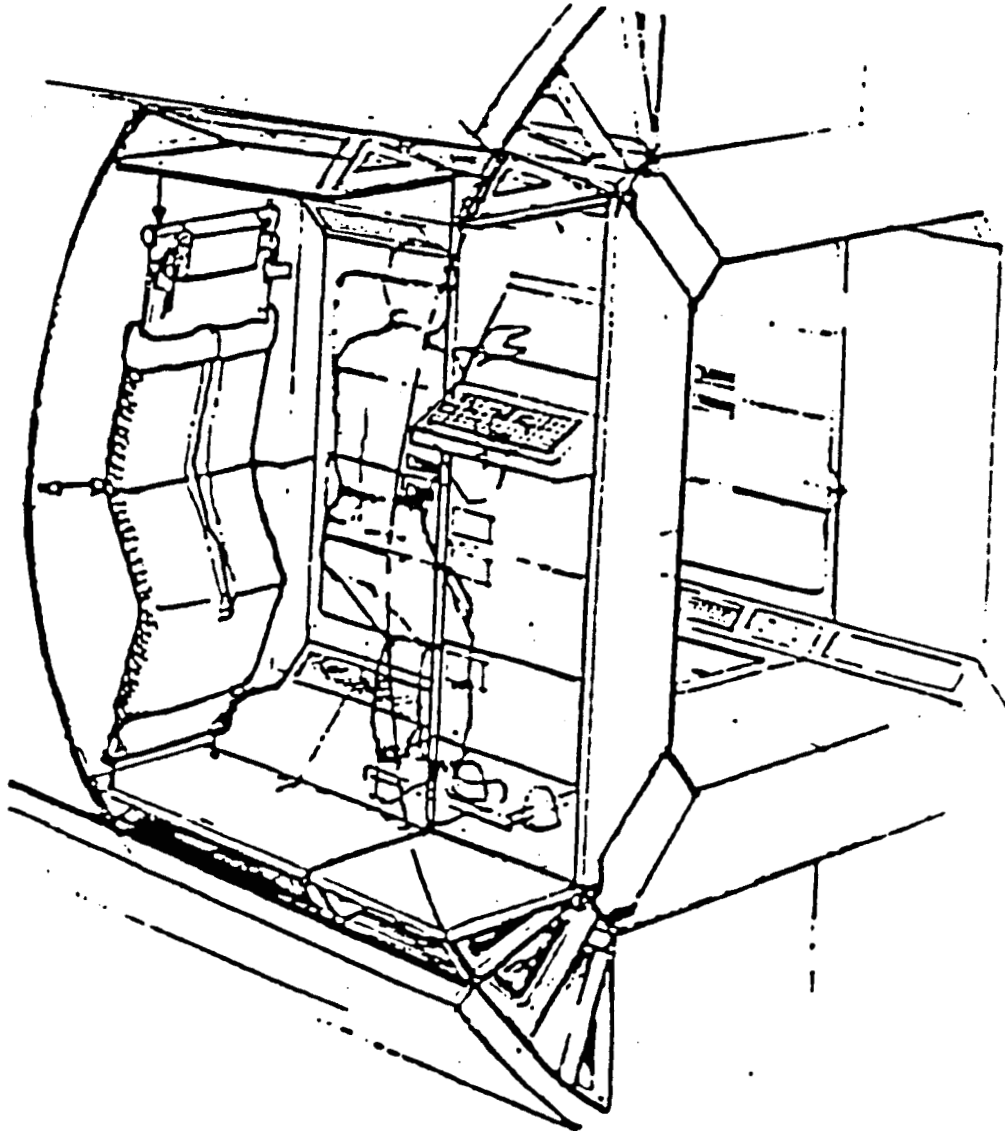
Figure 3-2
Crew Quarters

33

Wardroom

The wardroom provides space for the multi-use table and to make possible a large viewing window in the sidewall. The table provides space for the entire crew at one time for eating or conferencing, and is usable by the crew during off-duty time for recreation or conversation.

Galley

The galley unit contains frozen, refrigerated and ambient storage provisions for food (see Figure 3-3). The galley also houses the subsystems needed for preparing and serving meals, including combination microwave/convention ovens, hot and cold potable water/beverage dispenser, utensil stowage and pull-out counters. An interface with the data management system provides recipe and cooking instructions and automatic control of the various cooking facilities. Clean-up and housekeeping is supported by inclusion of a trash compactor and stowage, and a convenient hand washer (2:172).
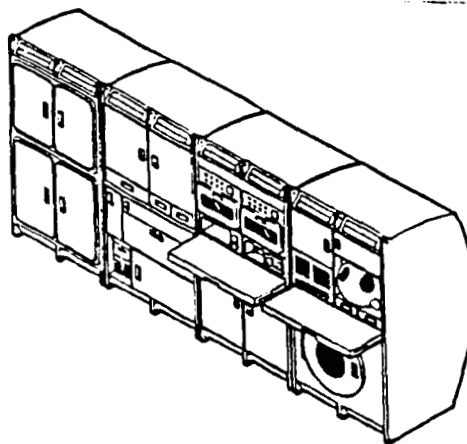
Figure 3-3
OTV Galley

Water Supplies

A human being requires approximately 15 lbs of water per day for both consumption and food preparation. It has also been estimated that 25 lbs of water per man per day is required for both personal hygiene and wash water (4). The necessary 120 lbs of water per day will be produced by the on-board power systems, (see Chapter IV, Power Systems).

Health Maintenance System

In zero gravity humans must exercise every day in order to maintain an appropriate level of physical fitness and slow the loss of minerals (especially calcium from the bones) and muscle deterioration. The OTV will use a treadmill type system which will provide resistance in the form of friction. Additionally, a bicycle ergometer will be provided to allow other muscle groups to be exercised.

Personal Hygiene Facility

The personal hygiene compartment is characterized by the requirement to house the maximum size crewperson within an enclosure that controls odor, contains accidental spills and provides privacy. This compartment contains facilities for shaving, oral hygiene, hand/partial body washing, personal grooming, etc., and has a backup urinal for use in the event the waste management compartment is occupied (2:172). There will not be a shower on board. A shower takes up too much room and presents difficulty in capturing all the globules of water.

Waste Management

Waste management will be handled by the Waste Management Compartment (WMC), a system similar to the one currently in use on board the Space Shuttle. The WMC is an integrated, multifunctional system, primarily utilized to collect and process biowastes from male and female crewmembers in a zero gravity environment (3). The system is used as a standard Earth-like facility. WMC performs the following general functions:

- Collects, stores, and dries fecal wastes and associated tissues

- Processes urine, and transfers it to the waste water tank

- Processes Extravehicular Mobility Unit (EMU) condensate from the airlock, and transfers it to the waste water tank

- Provides an interface for venting trash container gases overboard

- Provides an interface for dumping Air Revitalization System (ARS) waste water overboard in a contingency situation

- Processes wash water from the Personal Hygiene Station (PHS) and transfers it to the waste water tank.

Command Module

The command module houses all of the command and control modules as well as the spacesuits and other necessary equipment needed for EVA operations. The module was designed so that the spacesuits could be donned in the main compartment and then both astronauts, if needed, could enter the airlock to exit

the OTV. Reducing the size of the airlock reduces the time and energy required to operate the airlock.
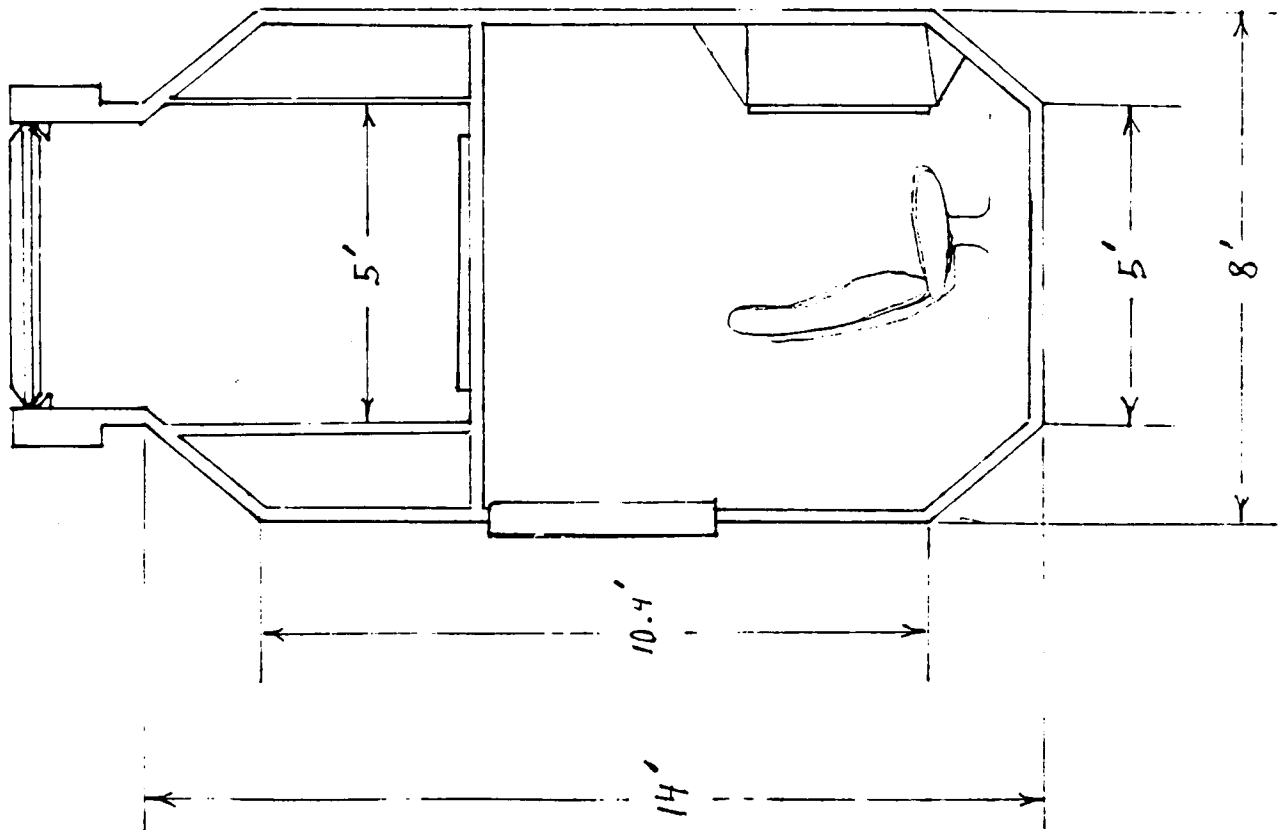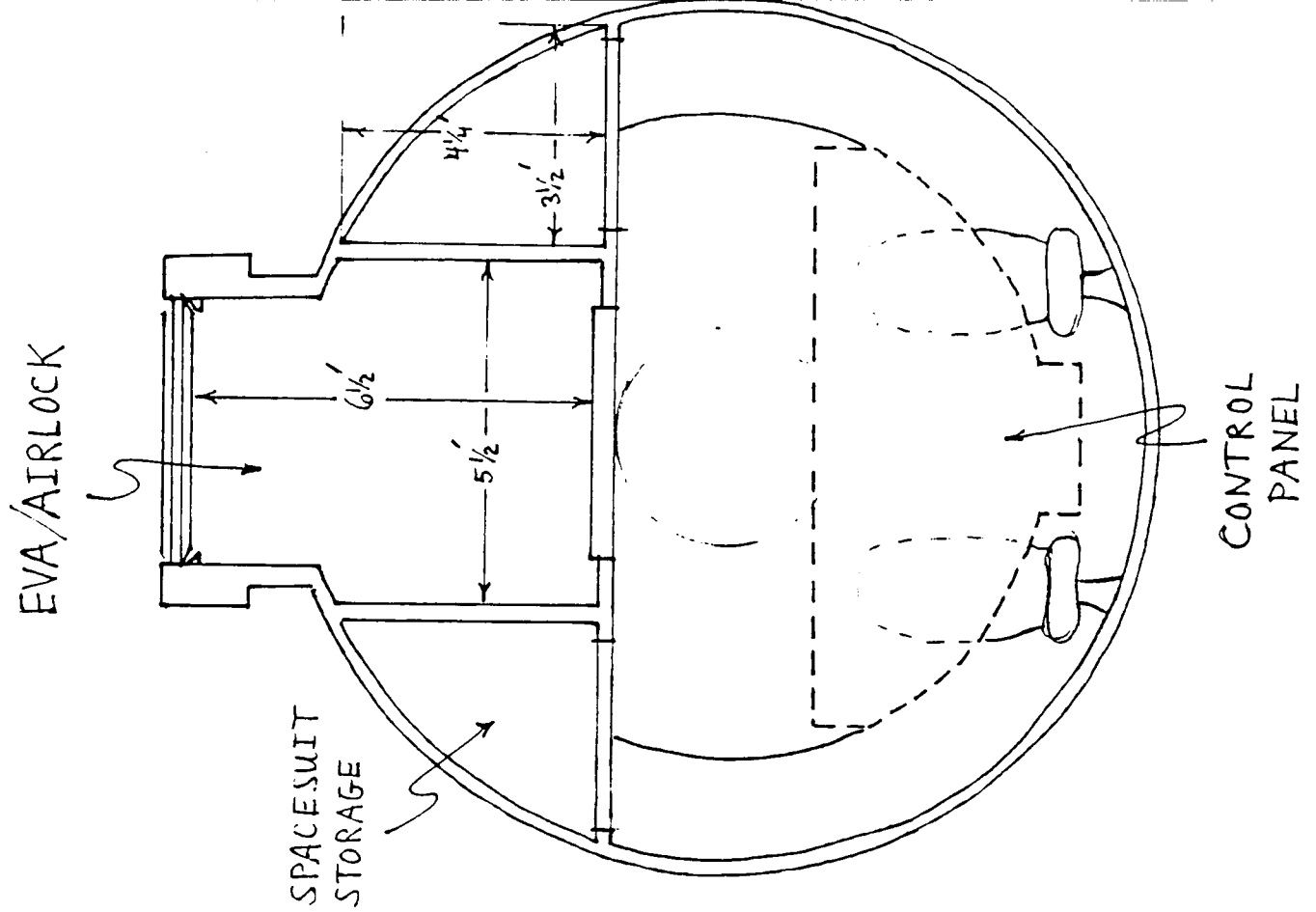
Figure 3-4.
Command Module

Conclusion

Though the mission is only planned to last a total of seven days, every effort should be made to make the astronauts stay in the OTV as comfortable as possible. It must be taken into consideration that the crew has already spent a significant amount of time in a stressful environment while living on the Space Station. Any unneeded hardship would only compromise the mission.

References

1. Chaikin, Andrew. "The Loneliness of the Long-Distance Astronaut." <u>Discover</u>. Vol. 6, Pgs 20-31. February, 1985.

2. Johnson, Gary L., et al. "Habitation Module for the Space Station." NASA Technical Report N86-40928. Huntington Beach, CA. 1986.

3. Lew, L. W. "Waste Collection System Workbook." NASA, Johnson Space Flight Center. WCS 2102C. November, 1984.

4. Puisson, S. H., et al. "MOSES - A Proposal for a Space Station Based Rescue Vehicle. University of Virginia. May, 1987.

5. Wise, James A. "The Quantitative Modelling of Human Spatial Habitability." NASA Technical Report N86-33210. Seattle, WA. 1985.

IV. ELECTRICAL POWER SYSTEM

Hydrogen-Oxygen Fuel Cell

Electrical power is a necessity on all modern space flights. The OTV is no exception to this. It needs power to run all its on board systems which include the environmental control/life support system, the communications and computer system, and the robot arm. There are a number of ways to produce electrical power in space but at the present time there is no single best way to produce electrical power in space. A number of systems were investigated for their potential as power systems for the Orbiting Transfer Vehicle. The systems investigated include solar, nuclear, and chemical power generation systems. A comparison was made between the three systems, and it was found that for the mission requirements and power needed by this vehicle, the best type of power system would be a chemical power generation system, specifically the use of fuel cells to produce the required power.

A hydrogen-oxygen fuel cell system has been used on manned missions starting with Gemini and Apollo up through the system currently being used on the space shuttle. It is; therefore, a proven system which can be installed with confidence on the OTV. It operates by using a chemical reaction of H_2-O_2 to produce power. Its weight is superior to all other systems for a mission which lasts only for a 50-200 hour duration. The fuel cell is easy to maintain and uses fuels that will be readily available in this Orbital Transfer Vehicle design. Because of its proven experience, compact size, and low weight this system was chosen
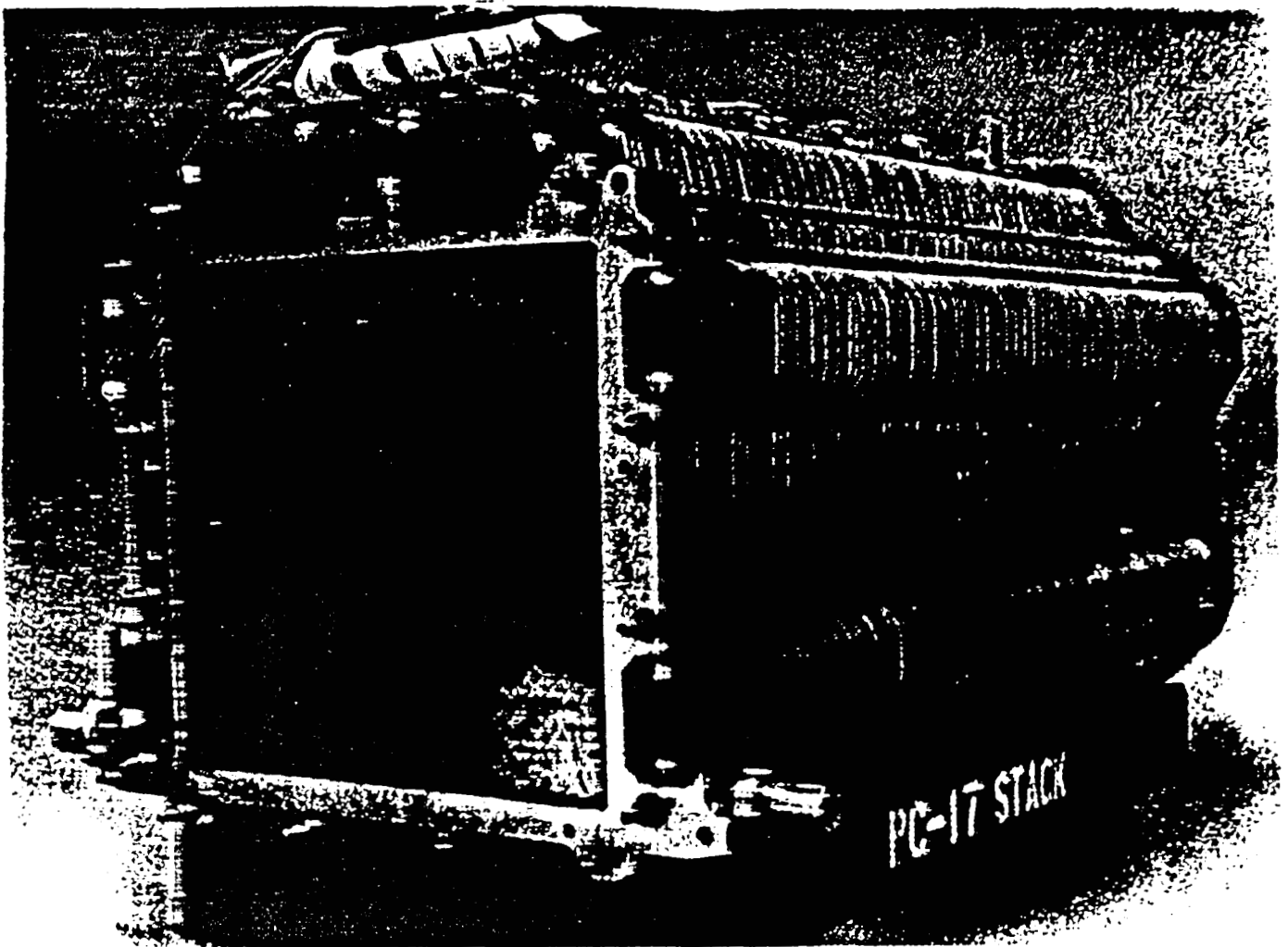
3 Substack Hydrogen Oxygen Fuel Cell

for the OTV.

The Space Shuttle, as mention above, currently uses an H_2-O_2 fuel cell system. The cells used on the shuttle are similar to the ones that will be used on the proposed OTV. The OTV will use two fuel cells. These fuel cells will contain three substacks of 32 subcells each. These subcells are where the power of the system is produced. There will be two cells operating continuously on the OTV so that if one cell becomes damaged the

enough power to return to the space station. In emergency situations when both fuel cells become inoperative, it would be possible to use the nuclear reactor, which is used as a propulsion system for the spacecraft, as a power source until the fuel cells could be repaired. Listed below are the specifications of the fuel cell to be used.

$$H_2-O_2 \text{ Fuel Cell Powerplant}$$

Type:	3 sub 32 cell system
Weight:	255 lb (cell), 2112 lb system
Dimensions:	45 in long x 15 in wide x 14 in high
Volume:	5.47 cubic feet
Tanks:	
Oxygen	Contain 781 lb of O_2 at 100-1050 psia and -260 to 170 F
Hydrogen	Contain 92 lb of H_2 at 100-355 psia and -402 to 170 F
Power:	12 KW at 27.5 VDC (normally)
	16 KW at 26.5 VDC (emergency)
Heat Rejection:	25 000 BTU/hour at 12 KW power produced
Water Produced:	10.7 pph at 12 KW power produced
Loads:	Can drive loads of 28.5 volts and 250 amps

Fuel Cell Operation

This system is easy to start up using a nickel-cadmium battery or the space station's own power supply to give the craft

the initial charge that it needs to start up the fuel cells, and it is easy to stop which involves only purging the system with water which the system produces itself. It integrates well with the Environmental Control/Life Support (ECLS) system by producing water which can be used by the ECLS system. It is estimate that the cells will produce 96 lbs of water per day for the crew an thermal control usage. It also uses oxygen which is needed in the ELCS system thereby not adding an extra tank to be used separately by the fuel cell. The heat, it rejects, can be eliminate by placing the cells on the strut supports of the OTV and by regulating their heat loss to the required level of operation of the cell.

Currently the amount of power needed by the OTV is listed below.

Power Requirements

Avionics:	1900 watts (min), 2361 watts (max)
Navigation:	800 watts (normal)
Crew Systems:	2700 watts (min), 2750 watts (max)
Docking Equipment:	2200 watts (normal)
ECLS system:	4000 watts (normal)
Robot Arm and EVA:	3750 watts (normal)
Total:	15350 watts (normal), 15861 watts (max)

The fuel cell system proposed will be able to handle these power requirements. A schematic of the system is shown in Figure 4-1.

The fuel cell system was chosen from among 3 other systems that were researched to determine which was compatible with the mission requirements given. These systems were solar, nuclear

Fuel Cell Powerplant Schematic
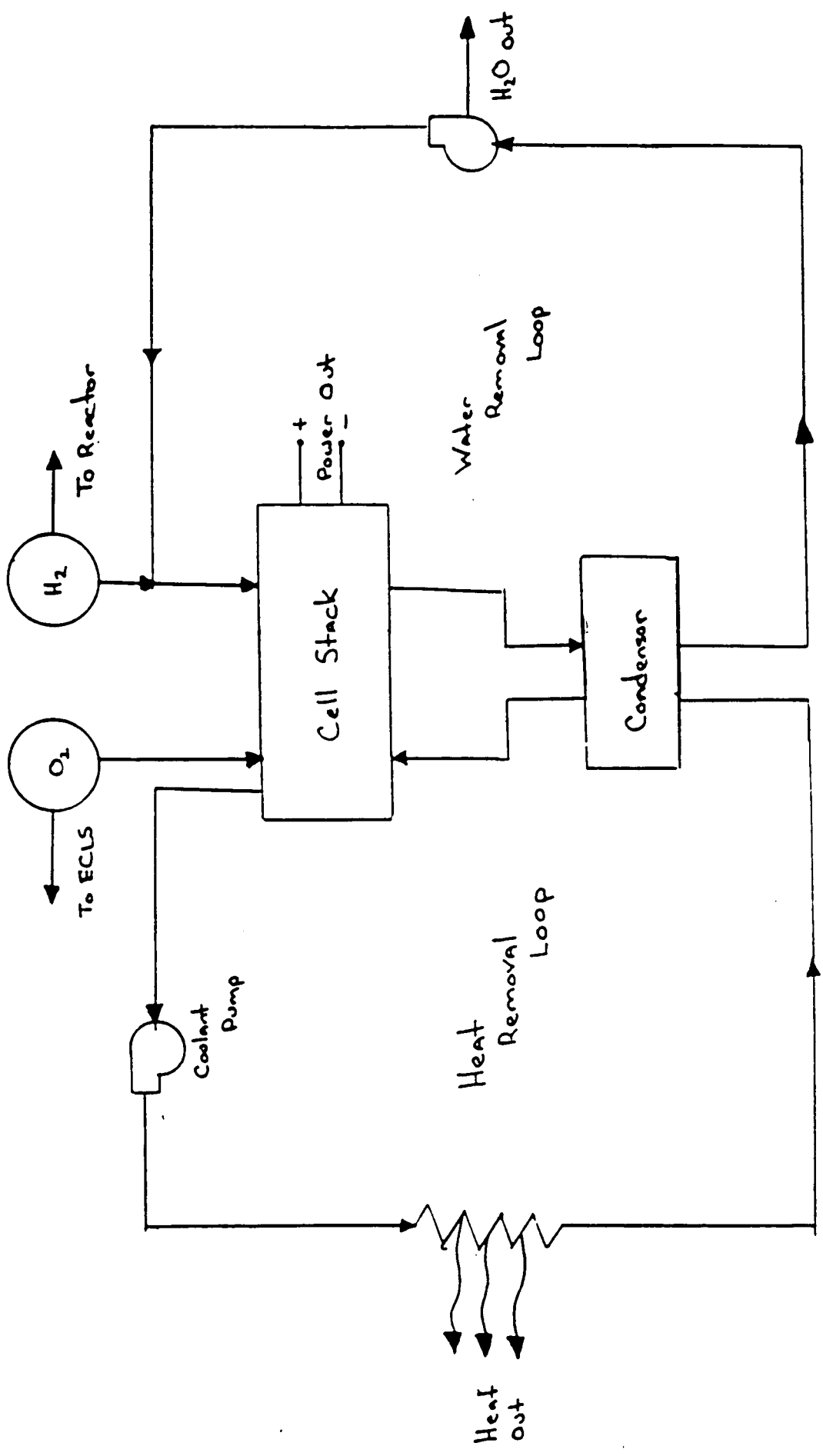
Fuel Cell Powerplant Schematic

Figure 4-1

and batteries. The problems with using solar power on an OTV include its size being to great and the need to continually adjust the panel so that it points at the sun to produce power. Nuclear generation is too dangerous on such a short mission. Its main benefit is that it produces great amounts of power when compared to its weight for longer periods of operation. Batteries were found to produce little power compared to the mission requirement, an if they were used the amount of batteries needed would far exceed the allowable weight limit. Figure 4-2 shows a comparison between the four power systems and the optimum mission length for each. The OTV mission length can be seen to fall in the fuel cells best operation condition. The technology currently existing for this power generation system, the amount of power that it can generate for the duration of the mission compared to its weight and its integration in the ECLS system make the Hydrogen-Oxygen fuel cell an attractive system. It is safe and well tried. This system will therefore be used on this OTV design.

POWER SYSTEM TRADEOFFS
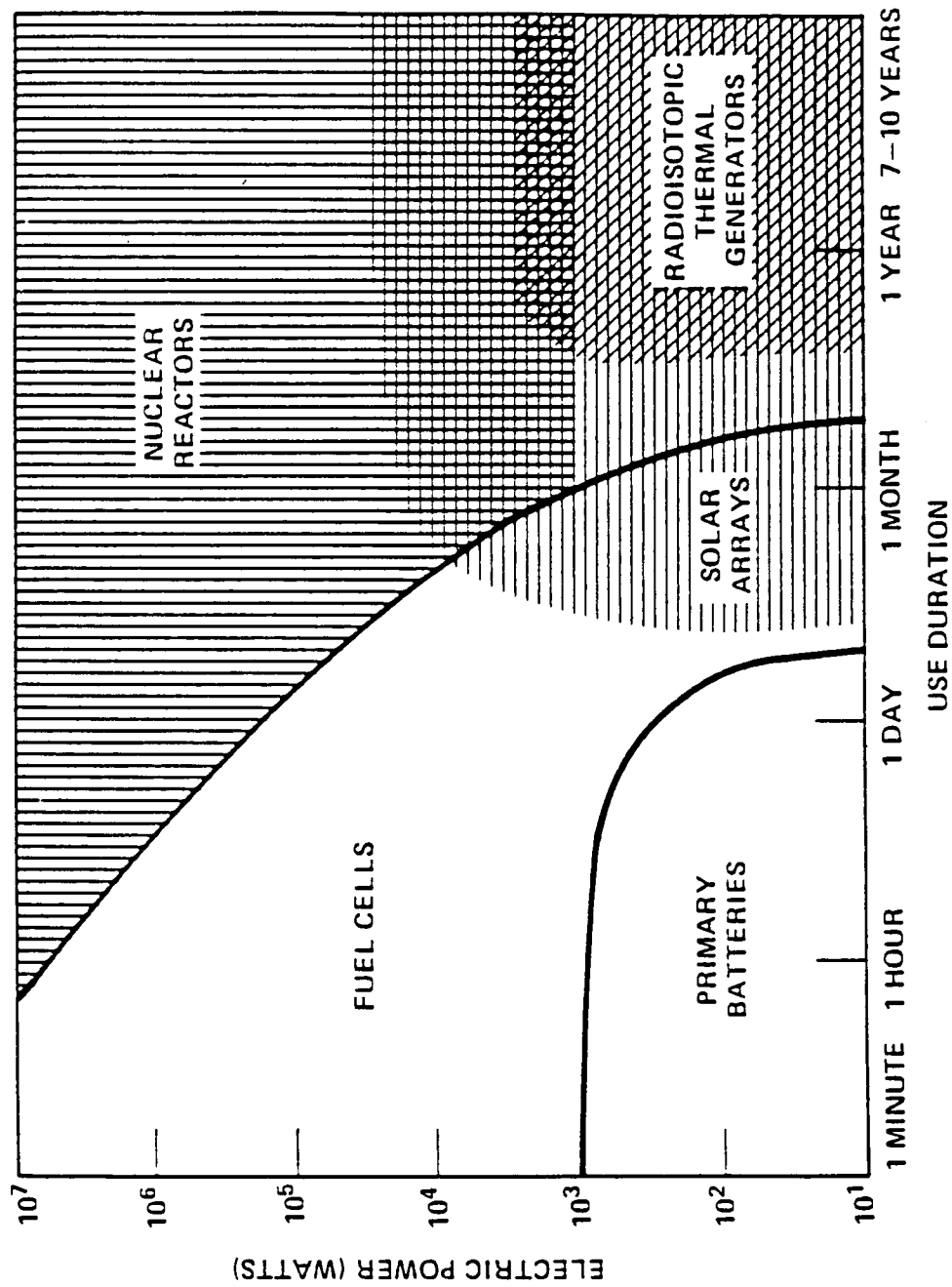
Figure 4-2

References

1. Corbett, Robert. <u>Power</u>. Lockheed Missiles and Space Company Report(N84-18288). Sunnyvale, CA:1984.

2. Faymon, Karl A. <u>Space Power Technology into the 21st Century</u>. NASA Technical Report, Lewis Research Center. Cleveland, OH:1984.

3. Fordyee, J. Stuart and Schwartz, Harvey J. <u>The Potential Impact of New Power System Technology on the Design of a Manned Space Station</u>. NASA Technical Report, Lewis Research Center. Cleveland, OH:1984.

4. Fordyee, J. Stuart. <u>Technology Status-Batteries and Fuel Cells</u>. NASA Technical Report, Lewis Research Center(N79-10132). Cleveland, OH:1979.

5. Gitlow, B. <u>Final Report-Development of the Advanced Fuel Cell System</u>. NASA Technical Report, Lewis Research Center(N79-12553). Cleveland, OH:1979.

6. Johnson, Richard E. <u>Orbiter Fuel Cell Improvement Assessment</u>. Le Tourneau College Technical Report(N82-19610). Longview, TX:1987.

7. Martin, R.E. <u>Topical Report - Electrochemical Energy Storage for an Orbiting Space Station</u>. NASA Technical Report, Lewis Research Center(N82-17607). Cleveland, OH:1982.

8. McBryar, Hoyt. <u>Technology Status - Fuel Cells and Electrolysis Cells</u>. NASA Technical report, Johnson Space Center(N79-10133). Houston, TX:1979.

9. NASA Training Manual. <u>Orbiter Fuel Cell Power Plant Review and Training Course</u>. South Windsor, CT: United Technology Power Systems, 1983

10. <u>Orbiter Fuel Cell Performance Constraints</u>. NASA Technical Report, Johnson Space Center(N80-19610): Houston TX:1980.

11. <u>Regenerative Fuel Cells - Technology Report</u>. NASA Research Report. 1984.

12. Sheilby, Dean W. <u>Regenerative Hydrogen-Oxygen Fuel Cell-Electrolyzer Systems for Orbital Energy Storage</u>. NASA Technical Report, Lewis Research Center(N84-33670). Cleveland, OH:1984.

and Reactant Supply System. NASA Technical Report, Johnson
Space Center(N85-16947). Houston, TX:1983.

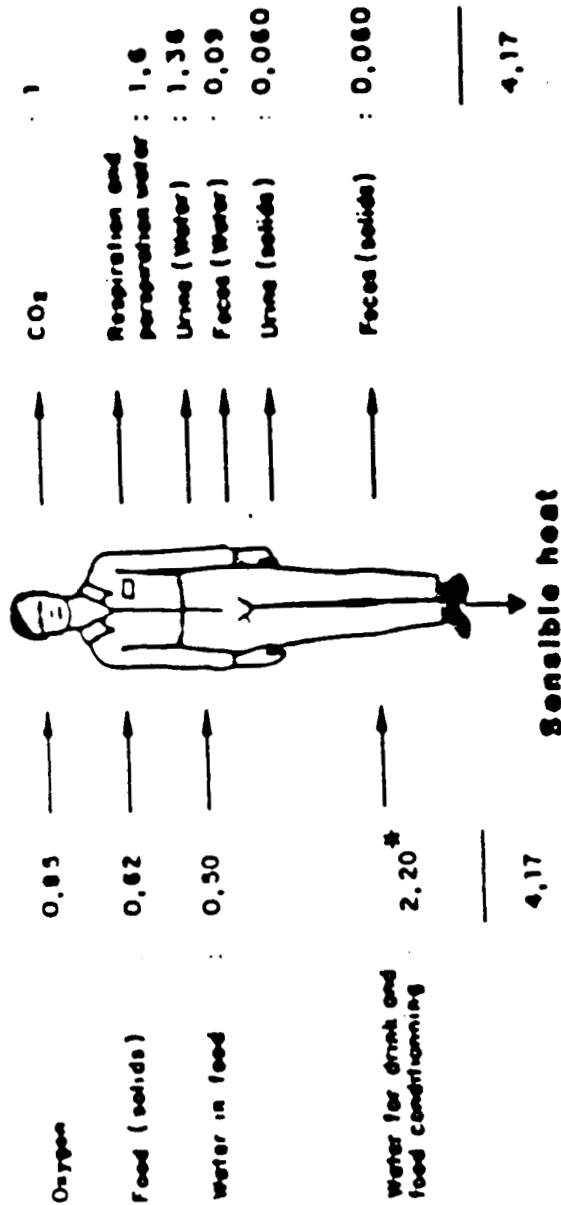# V. ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM

The environmental control/life support system is an important concept which must be addressed on any manned space mission. It will provide the crew with its daily life support needs as shown in Figure 5-1. There are six major areas of concern in crew life support which will be addressed in this report. They are :

1) Atmospheric Revitalization

2) Life Support

3) Water Processing

4) Active Thermal Control

5) Fire Protection

6) Air Lock Support

Figure 5-2 shows what units are defined under each area. This portion of the report will deal mainly with the first five areas while the sixth area will be explored in a later section of this OTV report. We will begin with a discussion of how closed the environmental control/life support system will be on the OTV.

The system that will be used on the OTV will be a partially closed system. From Chart 5-3, the weight of the system can be minimized by using a partial water recovery and a carbon dioxide removal system. A system which is closed more than this one will require more power than is necessary on the trip and will create weight problems. The closed system which will be used will decrease some of the launch weight/volume requirements, but the power required by the craft will also increase slightly over a totally open system. It is important to minimize the weight so that the amount of fuel needed to be carried is reduced and

Units: Kilogram/man x day (Kg/m x d)

Input		Output	
Oxygen	0.85	CO_2	1
Food (solid)	0.62	Respiration and perspiration water	1.6
water in food	0.50	Urine (Water)	1.38
		Feces (Water)	0.09
water for drink and food conditioning	2.20*	Urine (solids)	0.060
		Feces (solids)	0.060
	4.17		4.17

Sensible heat

Metabolic water production: 0.35 Kg/mxd

*2.5 Kg/mxd for long missions

HYGIENE: Depending on energy production system and mission duration

- Minimum (Short mission):0.3 Kg/mxd
- Maximum (Shower):5 Kg/mxd

Figure 5-1

The Life Support Needs of a Crew Member per Day

Taken From: A. I. Skoog, BLSS: A Contribution to Future Life Support. Dornier System Gmbtt Report(N85-29534). Friedrichshafen, Germany:1985.
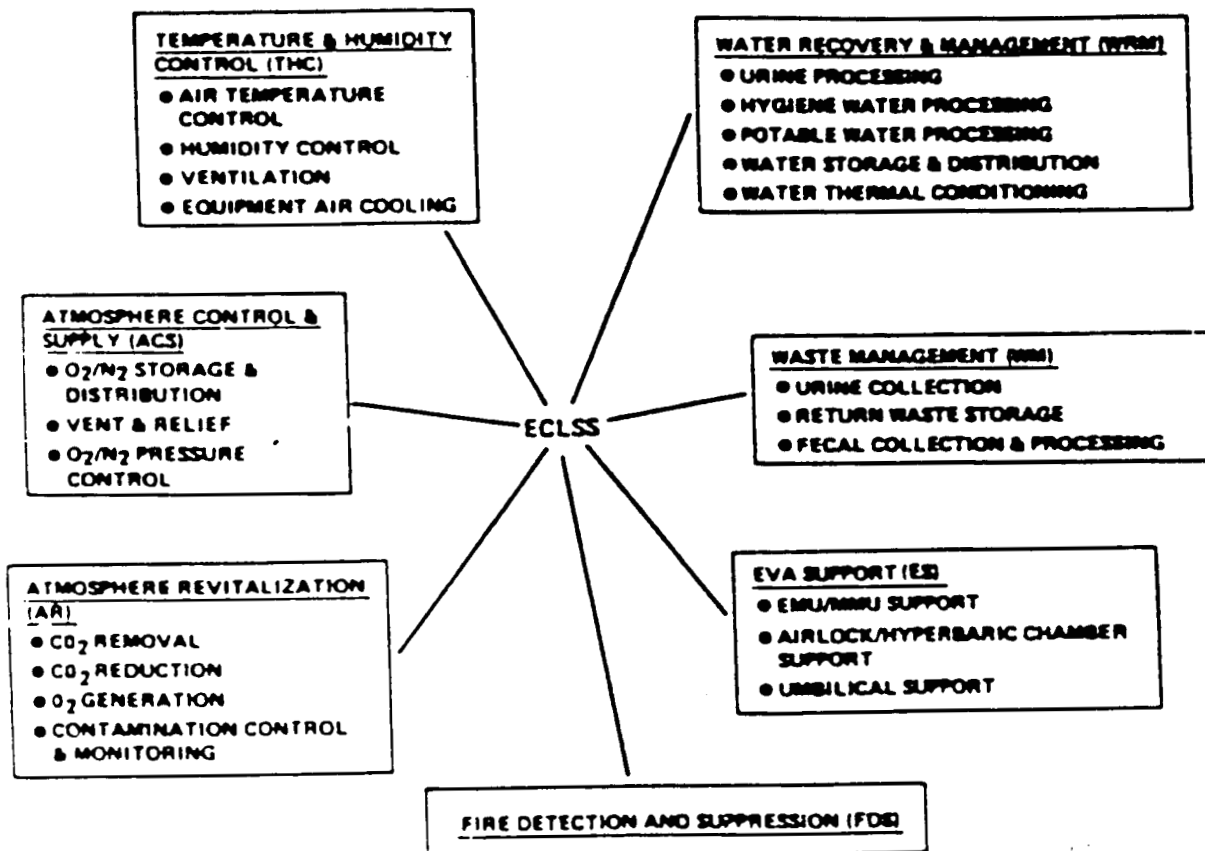
Figure 5-2

ECLSS subsystems

Taken From: Silvano Colombano, _Control Problems in Autonomous Life Support Systems_. NASA Technical Report, Ames Research Center(N83-30020). Moffett Field, CA:1983.

EQUIVALENT WEIGHT, lb × 10³

OPEN LOOP

PARTIAL WATER RECOVERY

WATER RECOVERY

CO_2 REMOVAL

O_2 GEN.

N_2 SUPPLY

CO_2 REDUCTION

MISSION DURATION, person-days
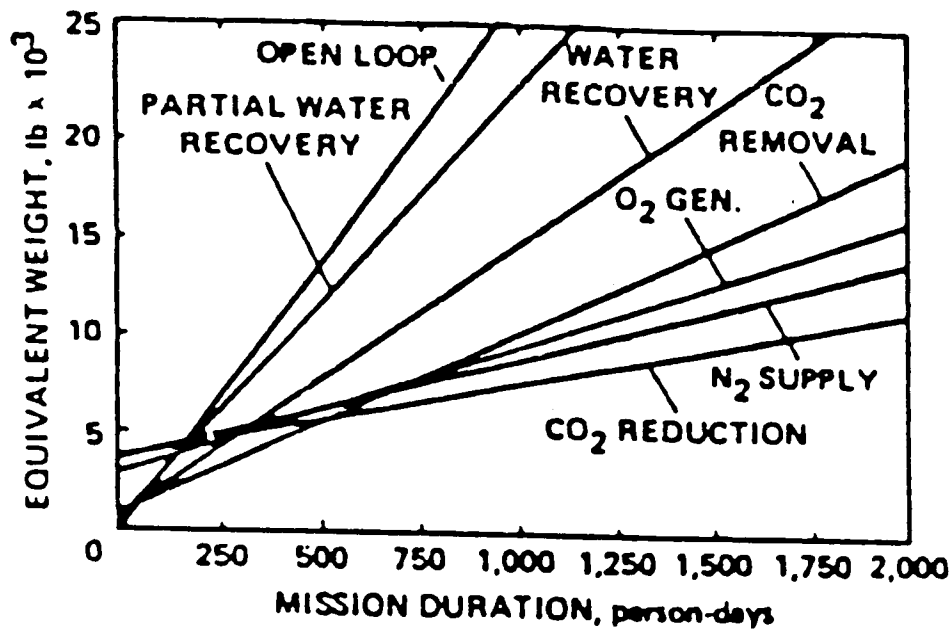
SEQUENTIAL STEPS IN LOOP CLOSURE

DEFINITION	DESCRIPTION
PARTIAL WATER RECOVERY	HUMIDITY CONDENSATE COLLECTION
WATER RECOVERY	POTABLE WATER RECOVERY AND TREATMENT FROM URINE AND WASH WATER
CO_2 REMOVAL (NON-EXPENDABLE)	REPLACEMENT OF EXPENDABLE LiOH WITH REGENERATIVE CO_2 COLLECTION TECHNIQUE
O_2 GENERATION	GENERATION OF O_2 THROUGH WATER ELECTROLYSIS USING RECLAIMED WATER
N_2 GENERATION	GENERATION OF N_2 THROUGH DISSOCIATION OF HYDRAZINE
CO_2 REDUCTION	DECREASE IN EXPENDABLE WATER BY RECOVERING PRODUCT FROM CO_2 REDUCTION (SABATIER) PROCESS

Chart 5-3

Mission Durations effect on Spacecraft Weight

Taken From: Concept Definition for an Extended Duration Orbiter ECLSS. Hamilton Standard Report(N79-23666). Windsor Locks, CT:1979.

therefore the cost of each trip is reduced. One of the important ways to reduce weight, as illustrated in Chart 5-3, is to close the carbon-dioxide loop, or the atmospheric revitalization system.

1. Atmospheric Revitalization

The atmospheric revitalization system controls the quality of the spacecraft's atmosphere. This portion of the system deals with trace contaminant control and carbon-dioxide reduction on board the craft. This involves the removal of CO_2, humidity control, dust and contaminant control. A trace contaminant sorbent bed will be employed to remove atmospheric contaminants other than CO_2 from the cabin. The beds are made of activated carbon. They are contained in a canister which draws in, through the use of a fan, the air in the cabin. The carbon then acts as a filter and cleans the air as it goes through and re-emits the clean air back to the cabin. Each canister weighs 16.7 lbs when full and lasts for 15 days so it would have to be replaced every 1.5 to 2 missions. These canisters are designed to remove trace contaminants only.

The crew in the cabin produces carbon dioxide, a contaminant that cannot be removed by the sorbent bed system. Through research it was found that CO_2 can be removed by a generally lighter and less volume system concept than is being used currently. LiOH cartridges which operate similar to the sorbent bed are being used to remove carbon dioxide. The sorbent bed system to be used on the OTV will be a regenerative type system

56

where the principle of combining CO_2 and H_2 to form water for use by the crew will be used to save weight. The process used will be the Sabatuer and Bosch processes. The CO_2 is converted to water via hydrogenation. The water will be used to control cabin temperature and humidity control and as a supplement for the potable water supply.

2. ECLSS Supply

The system has certain requirements which it must meet in order for it to provide adequate life support for the crew, and these requirements are shown in Table 5-4. The object of the environmental control/life support system (ECLS) is to provide an atmosphere as similar to earth's as possible. It is, therefore, designed to provide an oxygen/nitrogen mixture at 14.7 psi and has to be able to operate at a level of 8 psi without damage. The system will provide a cabin temperature of 70° F and be capable of withstanding a level of 10733 Btu/man-day. The craft must carry or be able to produce 53 lbs of cryogenic nitrogen and 260 lbs of cryogenic oxygen. The system must provide 50 lbs of food per mission to support the crew.

The crew must have, in addition to the above elements, water to survive. Water will be produced for crew use through the integration of the ECLS system with the fuel cell system. The crew needs 690 lbs of water on a 10 day, 3 man mission. The cell will produce 10 lbs/hr of water per 12 KW of power produced by the cells. It was estimated that the fuel cells could produce 2400 lbs of water in 10 days, easily meeting the water

Table 5-4 ECLSS Average Design Load

Metabolic O_2	0.83 kg/man day
Leakage Air	2.27 kg/day total
EVA O_2	0.55 kg/8 hr EVA per man
Metabolic CO_2	1.00 kg/man day
Drink H_2O	1.86 kg/man day
Food preparation H_2O	0.72 kg/man day
Metabolic H_2O production	0.35 kg/man day
Clothing wash H_2O	12.47 kg/man day
Handwash H_2O	1.81 kg/man day
Shower H_2O	3.63 kg/man day
Perspiration and respiration H_2O	1.82 kg/man day
Urinal flush H_2O	0.49 kg/man day
Urine H_2O	1.50 kg/man day
Food solids	0.73 kg/man day
Food H_2O	0.45 kg/man day
Food packaging	0.45 kg/man day
Urine solids	0.06 kg/man day
Fecal solids	0.03 kg/man day
Sweat solids	0.02 kg/man day
Charcoal required	0.06 kg/man day
Metabolic sensible heat	2.05 kW-hr/man day
Hygiene Latent H_2O	0.44 kg/man day
Food preparation latent H_2O	0.03 kg/man day
Wash H_2O solids	0.44 percent
Shower/hand wash H_2O solids	0.12 percent

requirement of the crew. Further integration between the two systems will allow the fuel cell to provide oxygen for the craft's crew cabin, and hydrogen for the carbon dioxide reduction process.

3. Thermal Control and Waste Management

The thermal control and waste management in the craft are also important parts of the ECLS system. The thermal control system removes latent heat produced by various equipment installed in the craft. This is accomplished by circulating freon through tubing in the spacecraft to pick up the heat. The freon then takes the heat transferred in the cabin to radiators located outside the craft on the skin where they radiate the heat into space. In this way the cabin temperature is controlled to the 70° F temperature required in all current manned spaceflight. An area for further study in this design is the thermal control of the heat produced in the reactor. The idea currently under consideration for the OTV specified in this report is to have a closed hydrogen loop where liquid hydrogen is circulated through the nuclear reactor in the engine to remove the heat energy there caused by engine usage. The hydrogen is then circulated out of the reactor in tubing to a series of tubes which have freon tubes wrapped around them. The freon will then take the heat in the hydrogen to the radiators to be liberated into space and the hydrogen will be returned to its holding tank to be used as fuel for the spacecraft. A further investigation of this system and other thermal control systems is required to choose the correct

system for the OTV.

The waste system collects and stores liquid and solid waste produced by the crew. This waste will be stored in a tank which holds 210 man-days of waste and this tank will be dumped into space every 4-7 days. The loads expected to be encountered in the system are 9.9 lbs/day of urine and 7.65 lbs/day of wash water which will be produced by a 3-man crew.

4. Fire Detection and Suppression

In the area of fire protection, the craft will contain smoke and heat detectors to warn of fire or potential fire. Systems in fire areas will be able to be shut down quickly and a foam suppression system will be used in electrical systems. There will be both hand and automatic fire suppression equipment which will be used to control any fire situation.

This system was chosen over closed systems due to its initial weight to length of the mission. The ECLS system is made by modular design. New modules can be put in to enhance the performance of particular systems, for example putting a module in to increase the efficiency of the CO_2 regeneration. The system can, therefore, easily be updated to provide expanded service to the crew on any mission which will be undertaken. With this type of system, the OTV can be modified to be used well into the 21st century.

References

1. Breeding, R.E. and Brose, H.F. <u>Environmental Control and Life Support (ECLS)for the Space Station - No Single Answer</u>. AIAA Technical Report(A84-11754). Windsor Locks, CT:1984.

2. Brose, Harlan F. <u>Environmental Control and Life Support Design Optimization Approach</u>. United Technologies, Hamilton Standard Report(A84-24637). Windsor Locks, CT:1984.

3. Colombano, Silvano. <u>Control Problems in Autonomous Life Support Systems</u>. NASA Technical Report, Ames Research Center(N83-30020). Moffett Field, CA:1983.

4. <u>Concept Definition for an Extended Duration Orbiter ECLSS</u>. Hamilton Standard Report(N79-23666). Windsor Locks, CT:1979.

5. Gibb, J.W. <u>Orbiter Environmental Control and Life Support System</u>. Rockwell International Technical Report(N79-31283). Doeney, U.S.A.:1979.

6. Hueter, U. <u>Thermal/Environmental Control and Life Support System Requirements for Future Space Applications</u>. NASA Technical Report, Marshall Space Flight Center(N79-31030). Huntsville, AL:1979.

7. Humphries, William R., and Powell, Luther E. <u>Space Station Environmental Control and Life Support Systems Conceptual Studies</u>. NASA Technical Report, Marshall Space Flight Center(A86-15813). Huntsville AL:1985.

8. <u>Inorganic Analysis in a Controlled Ecological Life Support System</u>. NASA Research Paper(N84-17853). 1984.

9. Lemignen, L. <u>Hermes ECLSS: Review of General Requirements and Application to the Spaceplane Definition</u>. International Astronautical Report(A86-15817). Toulouse, France:1986.

10. MacElroy, Robert D. ed. <u>Controlled Ecological Life Support System - Life Support Systems in Space Travel</u>. NASA Conference Publication #2378 (N85-29531). Graz, Austria:1985.

11. Nitta, K. and Yamashita, M. <u>Basic Consideration on CELSS</u>. International Astronautical Federation Conference Report(A86-15819). Stockholm, Sweden:1985.

12. Quattrone, P.D. <u>Extended Mission Life Support Systems</u>. NASA Technical Report, Ames Research Center(N85-29547). Motten Field, CA:1981.

13. Schubert, F.H. <u>Advanced Regenerative Environmental Control and Life Support Systems: Air and Water Regeneration</u>. NASA Technical Report, Ames Research Center(N85-29535). Motten Field, CA: 1985.

14. Shuey, M.A. <u>A Projection of Future Manned Spacecraft Environmental Control Systems</u>. Hamilton Standard Report(N79-31031). Windsor Locks, CT:1979.

15. Skoog, A.I. <u>BLSS: A Contribution to Future Life Support</u>. Dornier System Gmbtt Report(N85-29534). Friedrichshafen, Germany:1985.

VI. NAVIGATION AND ORBITAL MANEUVERS

Navigation

This section of the report details equipment for the OTV's basic orbital maneuvers. The main navigation components include a GPS (Global Positioning System) receiver, two star tracker units, and a laser gyro IMU (Inertial Measurement Unit). The equipment is presented in the context of an actual OTV satellite service mission in order to appropriately describe their function.

The first stage of the OTV mission is the orbital transfer for rendezvous with the target satellite. The craft begins in an orbit with the space station, and ends in an orbit with the satellite. Several different pieces of equipment handle control of the craft while in orbit, as well as positioning for the transfer.

Attitude control is the most basic requirement for maneuvering a spacecraft. This concerns keeping the craft in level flight, on the correct course, and free from rotation. The IMU controls attitude control (Fig. 1). The OTV IMU's consist of laser gyroscopes and accelerometers, aligned along the principle axes of the craft (the body-fixed x-y-z axes), and they measure acceleration and inertial attitude. Laser gyroscopes are used over conventional gyroscopes because of their high degree of accuracy. The laser gyroscopes work as follows: A light beam from a laser is split into two beams, which are subsequently directed on two directions around a closed course defined by
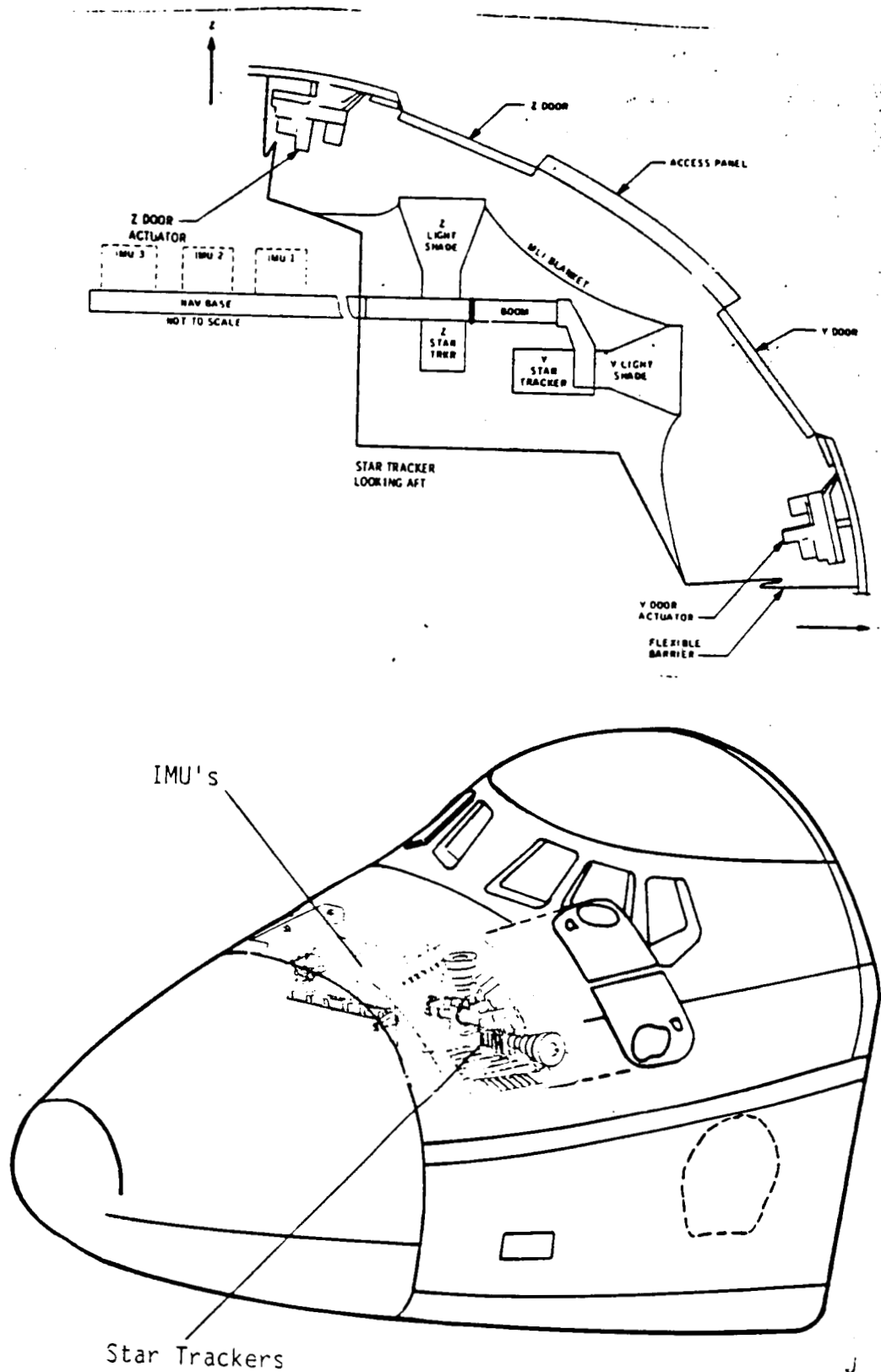
FIGURE 1.
The star trackers and main IMU units as
oriented in the space shuttle.

65

mirrors. The relativistic frequency difference between the two beams is a measure of the angular rate about the axis of the closed course. The accelerometers measure acceleration along the axes. The data provided by the gyroscopes and accelerometers is fed to the main IMU unit and to the Attitude Director Indicator (ADI), which displays roll, pitch, and yaw rates. This allows the pilot to correct the course of the craft.

The IMU measures change in attitude, but the attitude of the craft must be known. Also, the IMU units tend to drift over time, and so they are augmented by a second piece of navigation equipment: a position sensor of some type. These sensors are usually sun sensors, earth sensors, or star sensors, and they provide the orientation of the craft. Satellites may use any of these sensors, but a maneuvering craft is most likely to chose a star sensor device, referred to as star trackers. Star tracker units are simply sensitive light-receiving devices, like a camera, which record the positions of the different images they receive (Fig. 2). These devices are aligned in the spacecraft to record star positions. The OTV will employ two units, pointing at right angels to each other, and run them when attitude is required (Fig. 1). The positions of the stars are matched with a star catalogue contained in the main computer, known as the General Purpose Computer (GPC), thus establishing the orientation of the spacecraft. The OTV will employ the general purpose standard star tracker, which meets a wide variety of conditions while maintaining low weight and power requirements.

66

- **Electronics assembly.** Contains optics and electronics for converting optical images into digital data

- **Adapter plate.** Provides common mount for Star Tracker subsystem

- **Protective window.** Provides a seal that protects and allows electronics assembly to be pressurized

- **Light shade assembly.** Protects lens from glare. Also provides mount for Bright Object Sensor (BOS) and shutter mechanism.
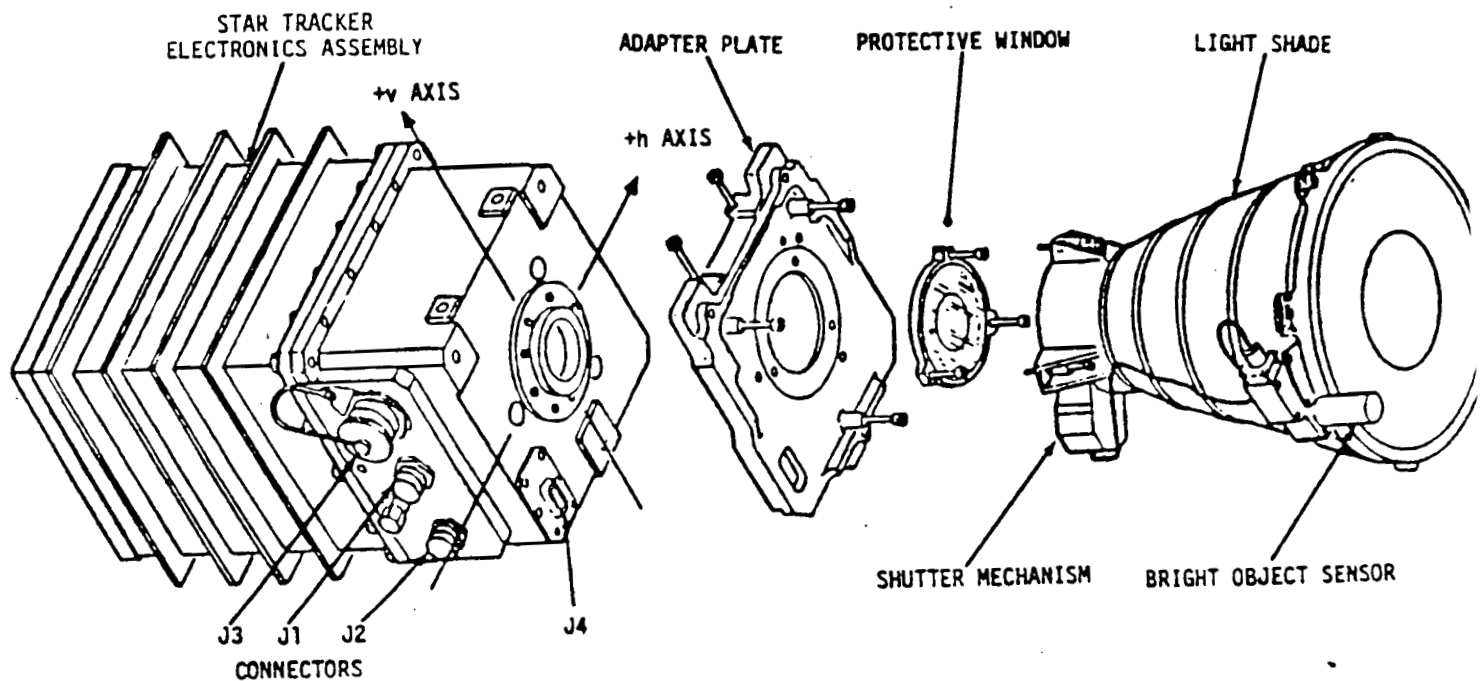

FIGURE 2.
A typical star tracker unit.

Specifications for the star tracker include:

1) 6th magnitude sensitivity.

2) Large 8°x8^{o} field of view.

3) Flight proven design on space shuttle.

4) 10 arc second accuracy.

5) -10°C to 50°C temperature range.

6) Withstands 20g acceleration.

The last piece of equipment used for orientation is a GPS processor/receiver. The GPS is a network of satellites and ground-based stations. The satellites will be placed into orbits with a period of 12 hours and inclined 55° from the ecliptic plane. There will be six of these orbits, each with three evenly spaced satellites (a total of 18), with each line of nodes spaced 60° from the next (Fig. 3). This means four satellites are in view from the ground at all times. The ground stations track these satellites, and send their position data to any craft with a receiver, including boats, airplanes, and spacecraft. The position data in synchronized so the receiving craft knows where the satellite was when it sent the signal. If the craft receives several signals from several satellites, its can determine its own position. Several of the GPS satellites are deployed, the rest await the continuation of the space shuttle program.
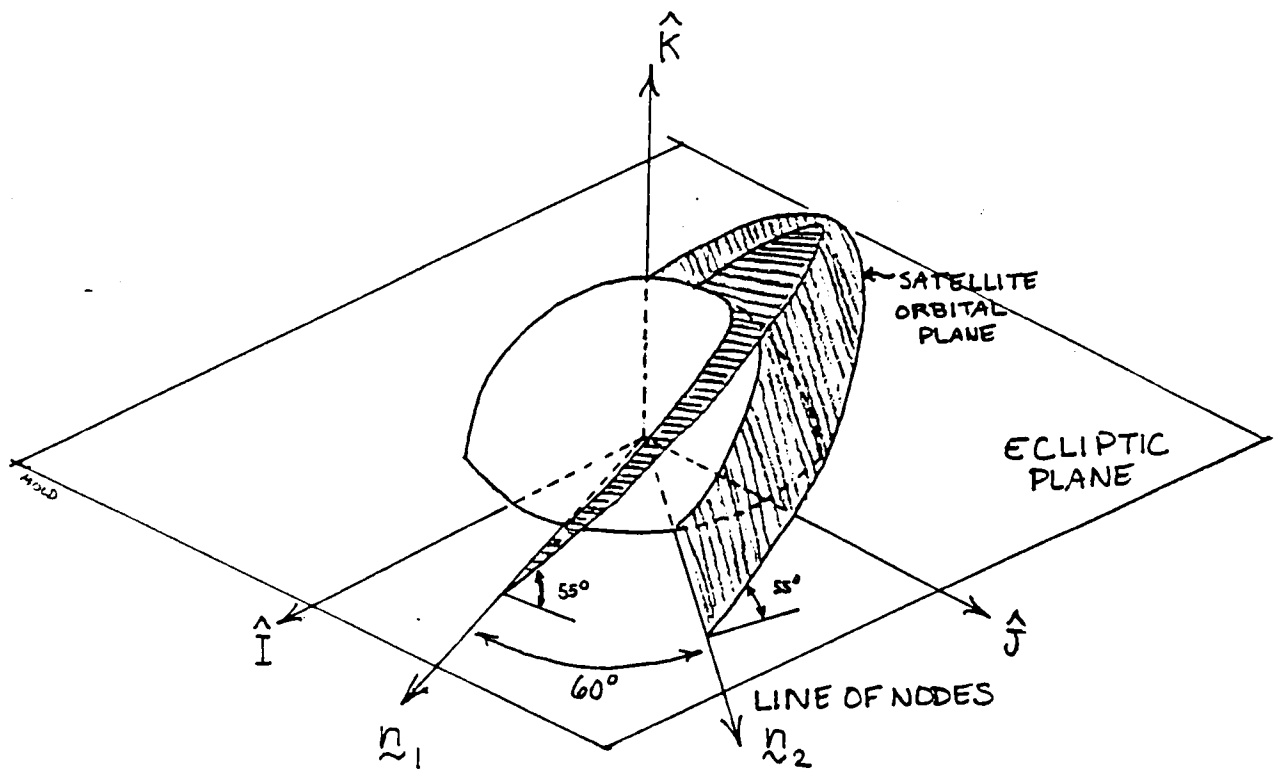
FIGURE 3.
The GPS system.

Proximity Operations

The other stage of the OTV mission begins when the OTV reaches the proximity of the target. This stage includes tracking and docking, and requires a different set of navigation equipment, specifically a Laser Docking Sensor, a laser ranger, and a high gain antenna.

69

Previously, the shuttle used its Ku-band Radar and Communications System for docking. However, this system is not sufficient for close-range operations because (1) it does not measure attitude, (2) it is not effective at ranges less than 100 feet (the specified range for the beginning of proximity operations), (3) it cannot perform radar and communications simultaneously while docking a station-keeping, and (4) it is too large and heavy for the OTV. The solution was to research the Laser Docking Sensor.

The OTV will employ the Laser Docking Sensor when the OTV comes within 100 feet of the target, presumably augmented by the laser ranger. The description of the laser sensor is an optical radar which uses a semi-conductor transmitter and an image-dissector receiver.

Specifications of the Laser Docking Sensor are:

1) It can augment or replace visual tracking of the target.

2) It supports both hard-docking (physical dock) and soft-docking.

3) It enables the OTV to determine relative position and relative attitude of the target.

4) It can perform long term station-keeping in an automatic mode to relieve the crew from continually monitoring the OTV position and apply corrective maneuvers.

5) It can tolerate viewing the sun without damage.

6) It is small and low-power.

70

The laser ranger and the high gain antenna complete the radar and distance equipment used for docking.

All of the state-of-the-art navigation equipment required by the OTV is summarized as follows:

Unit	Weight	Power	Volume
IMU (2)	36 lb	320 W	1100 in^3
Star Tracker (2)	36 lb	40 W	1150 in^3
GPS receiver (2)	40 lb	40 W	400 in^3
Laser Sensor	30 lb	50 W	6100 in^3
Laser Ranger	60 lb	350 W	2200 in^3
Totals	202 lb	800 W	10,950 in^3

Rendezvous

Rendezvous is the first stage of orbital maneuvers, namely, the orbital transfer. The OTV Earth orbital transfer is most interested in conserving fuel, and the transfer involving the least amount of fuel consumption is a Hohmann transfer. A Hohmann transfer is simply an ellipse drawn between two orbits (Fig. 4), with the earth at the focus, the perigee at the lower
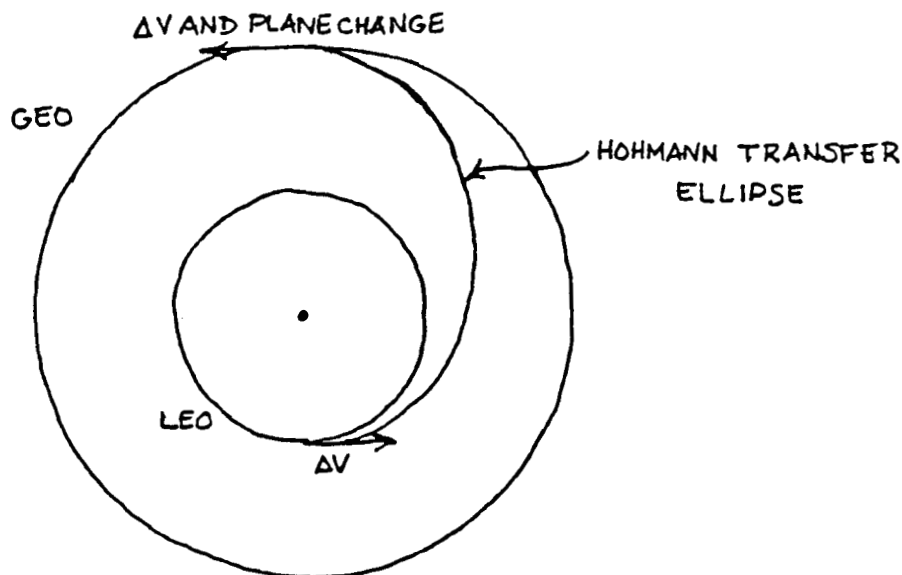
FIGURE 4.
A Hohmann transfer ellipse from low earth
orbit (LEO) to geosynchronous orbit (GEO).

orbit, and the apogee at the opposite side on the higher orbit.
The transfer follows one half of the ellipse. The OTV performs
an initial engine burn (expressed as $\triangle$ V) at perigee to place it
on an elliptic trajectory, and a second burn when the spacecraft
reaches apogee to circularize the orbit.

Also, since the space station on which the OTV is based is
inclined at 28.5° from the ecliptic and most satellites serviced
will be at 0° inclination (geosynchronous) or some other
inclination, the orbital transfer will involve a plane change.
The fuel requirement is lowest if the OTV performs the plane
change at the end of the elliptic transfer, and is even lower if
the second burn and plane change are done together.

Since in a Hohmann transfer the spacecraft finishes directly

72

opposite of where it started, is the time of flight for the transfer is calculated, then the position of the target when the transfer began can be obtained. Knowing both the initial position of the OTV and the target yields the initial conditions: in this case, the angle between both the target and the rendezvous point, and the OTV and the rendezvous point. Whenever these initial conditions are present, the spacecraft can begin an
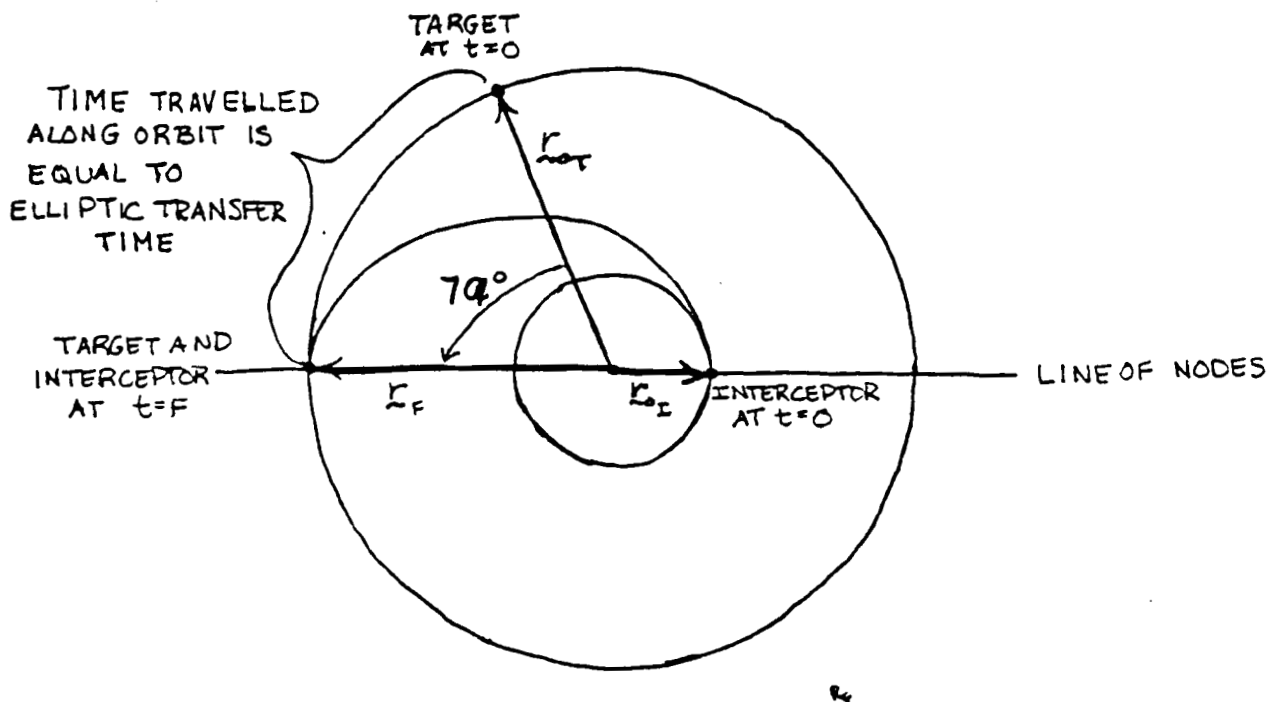

FIGURE 5.
The intercept problem.

73

orbital transfer. Since the time of flight is 5 hours, this requires that the target be 79° in front of the rendezvous point when the transfer begins (See Appendix A).

However, the final rendezvous can only take place on the line of nodes because of the plane change involved, so the OTV must begin the transfer when it crosses the line of nodes, and the target must be 79° away from crossing the line of nodes. This situation occurs very infrequently, if at all. However, a very simple maneuver will enable the spacecraft to perform the rendezvous every 12 hours, with no addition to the velocity increment and little addition to the time of flight.

The HITME (Hollo-Ibarra Transfer Maneuver Ellipse) maneuver, a new transfer technique originated by two of my peers, has the spacecraft initially placed into a small waiting ellipse by a $\triangle$ V, where it will return to perigee exactly when the target is 79° from the line of nodes, and the transfer can begin (Fig. 6). During one orbit of the OTV in LEO, the target travels 22.5°. The target will cross into the Initial Launch Zone (ILZ) once every 12 hours. Since the ILZ is 22.5°, at some time while the target is in the ILZ the OTV will be at the initial transfer point. Then the angle of the target between its position and the 79° mark is quickly obtained, along with the time of flight for the waiting ellipse, the parameters for the waiting ellipse, the initial velocity increment, the intermediate velocity increment, and the final velocity increment (See Appendix A). These calculations are not difficult and can be done by computer
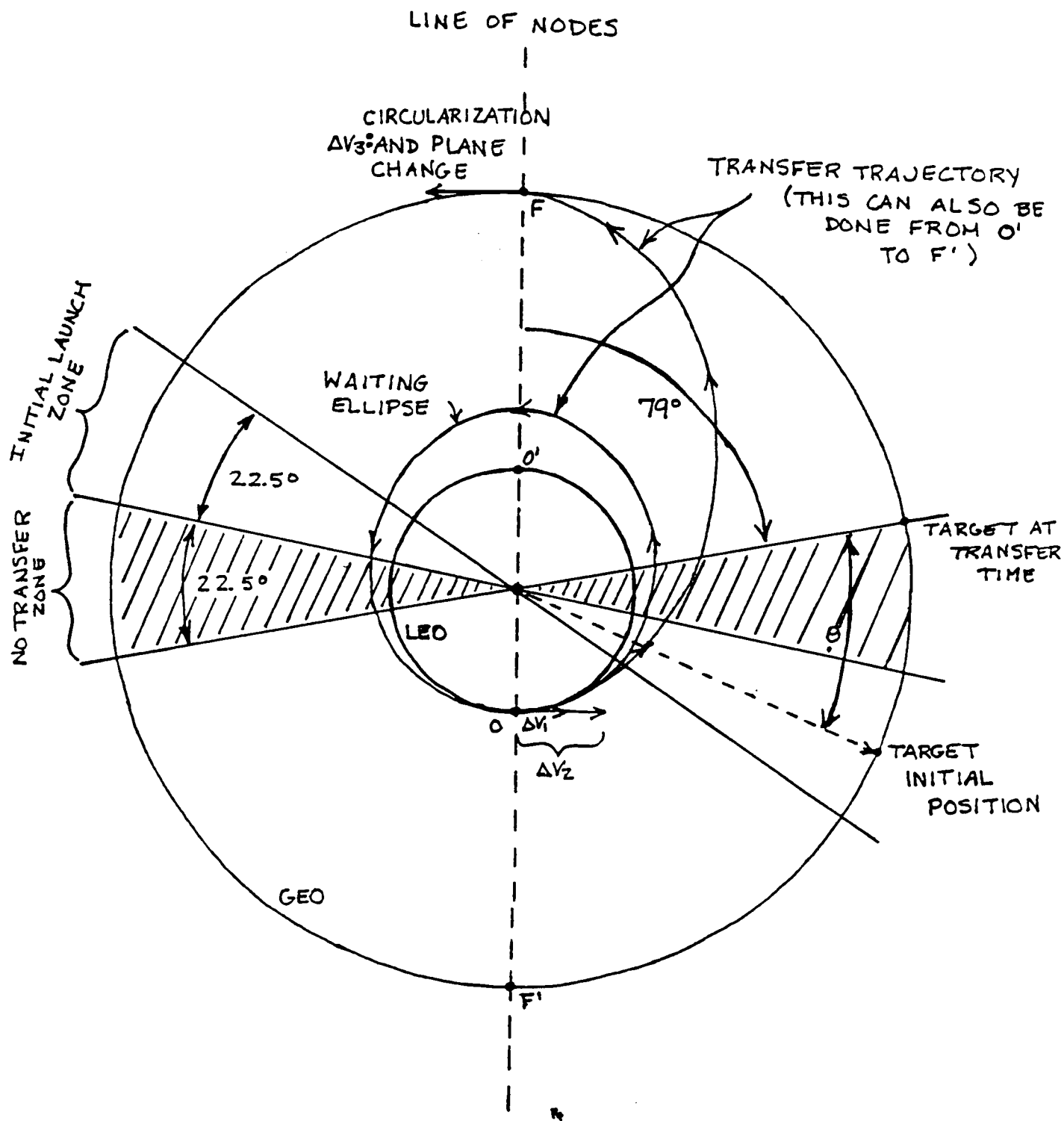
FIGURE 6.
The HITME maneuver.

(Appendix B).   Calculations show that this transfer maneuver, while increasing transfer time, adds no extra velocity increment, and still minimizes fuel consumption.

The No Transfer Zone (NTZ) is required between the ILZ and the 79° mark because if the OTV were to perform a waiting ellipse maneuver while the target was in the NTZ, the OTV would have to enter an elliptical orbit closer to earth in order to reach the initial transfer point, because the time of flight for the waiting ellipse would be less than one period of the OTV orbit, and that requires a lower orbit.

The return transfer is similarly done, except that no initial rendezvous angle is required (Fig. 7).   The OTV can depart any time it is on the line of nodes, performing the transfer to an elliptical orbit simultaneously with the plane change.   When the OTV reaches LEO, the angle of the space station from the rendezvous point is taken and the time to the rendezvous point with the OTV is calculated, adding on one complete orbit so that the new waiting ellipse is not performed below LEO.   The total time of flight of the space station reveals the waiting ellipse time of flight, and parameters are similarly calculated. These are also easily performed by computer (Appendix B).
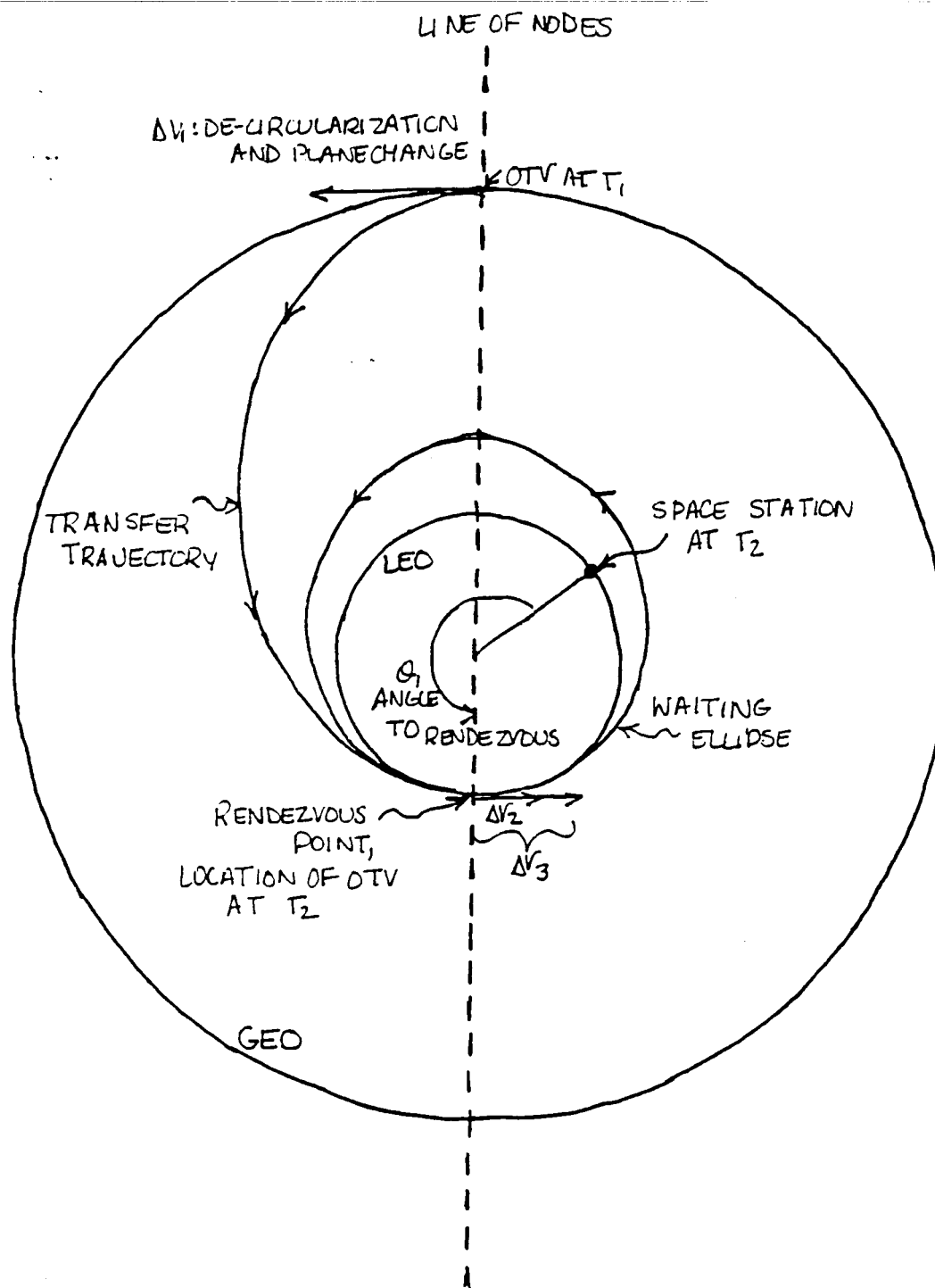
LINE OF NODES

ΔV_1: DE-CIRCULARIZATION AND PLANECHANGE

OTV AT T_1

TRANSFER TRAJECTORY

SPACE STATION AT T_2

LEO

θ_1 ANGLE TO RENDEZVOUS

WAITING ELLIPSE

RENDEZVOUS POINT, LOCATION OF OTV AT T_2

ΔV_2

ΔV_3

GEO

FIGURE 7.
The return maneuver.

Docking

Docking is the last phase of rendezvous. It has an initial range of 1500 to 100 feet. A common method of docking deals with a control cone, which has its vertex at the target (Fig. 8). When the OTV reaches the edge of the cone, velocity is reduced to the required range-rate (closure velocity). Whenever the OTV subsequently touches the boundary of the cone, an impulse thrust sends it back towards the middle. The boundaries of the cone are measured by the docking system, and the pilot reads the parameters (such as range-rate, angular rate, and displacement from the nominal line) and corrects the spacecraft attitude. As the cone narrows, the OTV closes on the target.
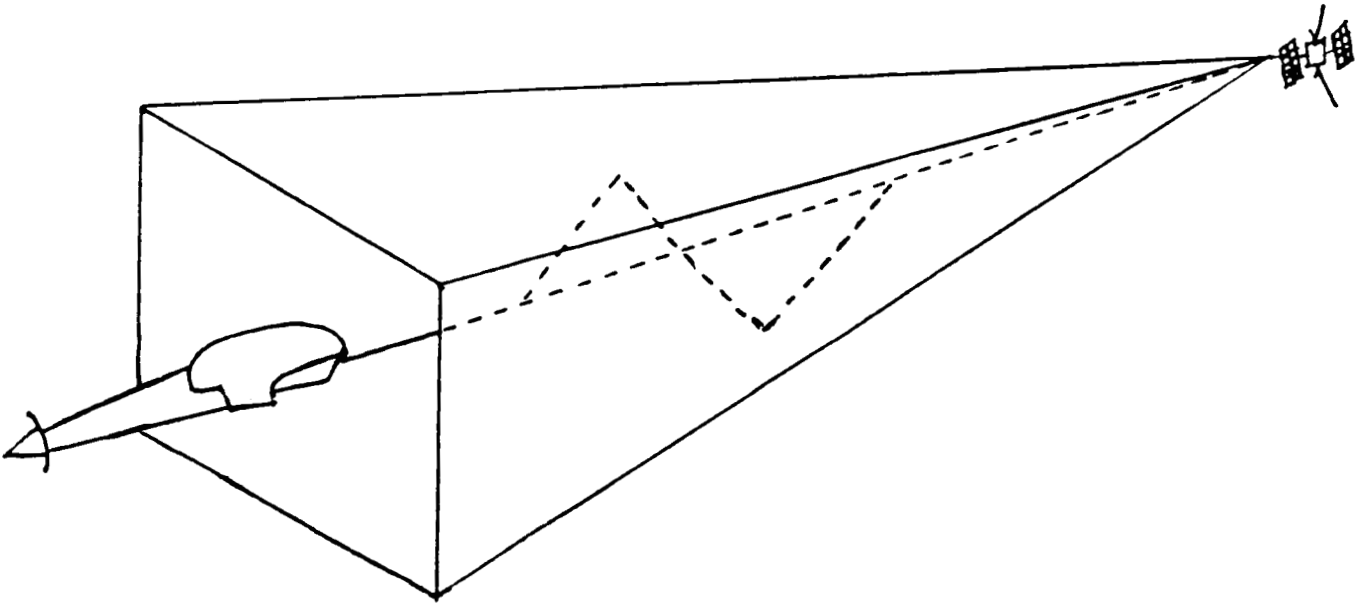

FIGURE 8.
Docking control cone.

78

BIBLIOGRAPHY

1. Bate, R. R., Mueller, D. D., and White, J. E., _Fundamentals of Astrodynamics_, Dover Publications, Inc., New York, 1971.

2. Class Notes, AE 425, H. S. Morton, Fall 1987.

3. Erwin, H., "Laser Docking Sensor", Satellite Services Workshop - Berthing/Docking, 1983, N83-11194.

4. _Future Orbital Transfer Vehicle_, Boeing Aerospace Company, Technology Study 1981, NASA-CR-3536, Vol. II.

5. _Guidance and Control Sensors_, GNC Systems Training Manual, NASA - Johnson Space Center, January 1985.

6. Kaplan, M. H., _Modern Spacecraft Dynamics and Control_, John Wiley & Sons, New York, 1976.

7. _Manned GEO Sortie Study_, NASA - Johnson Space Center, 1983.

8. _The NERO Project_, MIT Student Project, MIT Press, 1967.

9. _Star Tracker/ Crew Optical Alignment Sight Workbook_, NASA - Johnson Space Center, September 1983.

10. Sorenson, A. A., and Williams, I. J., "Spacecraft Attitude and Control", TRW Electronics & Defense Quest, Summer 1982.

APPENDIX A

The Rendezvous Maneuver

For this problem,

$$R_1 = 2.182 \times 10^7 \text{ ft}$$
$$R_2 = 1.382 \times 10^8 \text{ ft}$$
$$U_e = 1.4076 \times 10^{16} \text{ ft}^3/\text{s}^2$$

where R_1 is the radius of LEO, R_2 is the radius of GEO, and U_e is the Earth's gravitational parameter.

The orbital periods are

$$TP_1 = 5400 \text{ s or 1 hour, 30 minutes}$$
$$TP_2 = 86160 \text{ s or 23 hours, 56 minutes}$$

The transfer time for the Hohmann transfer is given as

$$TOF = \pi\sqrt{(a_t^3/U_e)}$$

where $a_t = (R_1 + R_2)/2$ and is the semi-major axis of the transfer ellipse. For the LEO to GEO transfer, TOF = 18950 s with $a_t = 8.00 \times 10^{17}$ ft.

In 18950 s the target travels

$$(360°/86160\text{s}) \times 18950\text{s} = 79.18°$$

This defines the angle that the target must have from the line of nodes when the OTV begins its transfer ellipse.
Also, during one orbit of LEO (5400 s) the target travels

$$(360°/86160\text{s}) \times 5400\text{s} = 22.55°$$

This defines the Initial Launch Zone for the initial waiting ellipse, and well as the No Transfer Zone.

To obtain the parameters for the waiting ellipse, as well as the $\triangle$ V's, the following calculations were performed:

First, obtain the 0 between the target and the 79° line. Then, find the time of flight for LEO corresponding to that angle.

$$TOF = \theta \times (5400\text{s}/360°)$$

Next, the elliptic parameters for that time of flight can be determined.

$$a_e = [(TOF/2\pi)^2 xU_e]^{1/3}$$

Where a_e is the semi-major axis of the wait ellipse. Since R_1 is the perigee radius, the eccentricity, e, can be determined from

$$e = 1-(R_1/a_e)$$

The semi-latus rectum, p, is found from

$$p = a_e x(1-e^2)$$

And h, the angular momentum per unit mass, is

$$h = \sqrt{pxU_e}$$

Now, the velocities are found form the following equations.

$$VP = h/R_1, \quad VP \text{ is the perigee velocity}$$

$$\Delta V_1 = VP - VC_1$$

ΔV_2 is the velocity increment required for the OTV to attain the transfer ellipse. V_i is the velocity at perigee for the transfer ellipse, or 33381 ft/s. Now, since VC = 6018 ft/s, or the combination circularization and plane change velocity increment, the total Δ V is given as

$$\Delta V = \Delta V_1 + \Delta V_2 + VC$$

The total time of flight is the sum of the TOF of the waiting ellipse plus the elliptic transfer time.

$$\text{Total transfer time} = TOF + 18950 \text{ s}$$

All of the above calculations can be programmed into a computer, and for each transfer and instant readout can be obtained. These calculations are performed by the computer program in appendix B. The return maneuver calculations are just as easily done, and are demonstrated by the second program in appendix B.

81

APPENDIX B

The Program for the transfer up

```
10 CLS
20 LPRINT" THETA      a (ft)          e          p (ft)    /\Vt(ft/s)   TOF(hr min
)
30 LPRINT"-------------------------------------------------------------
---
40 U=1.4076468B2D+16:R1=2.1B2E+07:R2=1.3B2E+08
50 VC1=25399:VC2=10092:V1=33381!:V2=5270
60 VPL=2595:DV3=4822:VC=6013
70 FOR TH=24 TO 45 STEP .5
80 TOF=TH*239
90 A=(((TOF/6.28)^2)*U)^.333
100 E=1-(R1/A)
110 P=A*(1-E^2)
120 H=SQR(P*U)
130 VP=H/R1
140 DV1=VP-VC1
150 DV2=V1-VP
160 DVT=DV1+DV2+VC
170 IF X=1 GOTO 200
180 TT=TOF+13950
190 GOSUB 1000
200 LPRINT USING " ##.#    #######    #.#####    ########    ##.##    #
  ##";TH;A;E;P;DVT;H;M
210 IF X=0 GOTO 230
220 LPRINT USING " ##.#          ####.#          ####.#          -###";
H,DV1,DV2,VC
230 NEXT TH
240 IF X=1 GOTO 300
250 LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT
:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT
260 LPRINT" THETA          /\V1(ft/s)          /\V2(ft/s)          /\V3(ft/s)
270 LPRINT"-------------------------------------------------------------
280 X=1
290 GOTO 40
300 END
1000 H=FIX(TT/3600)
1005 M=((TT/3600)-H)*60
1020 RETURN
```

Results of the program

THETA	a (ft)	e	p (ft)	ΔVt(ft/s)	TOF(hr min)	
24.0	22348150	0.02363	22335650	14000	6	51
24.5	22657110	0.03695	22626190	14000	6	53
25.0	22964040	0.04982	22907050	14000	6	55
25.5	23268860	0.06227	23179650	14000	6	57
26.0	23571770	0.07432	23441590	14000	6	59
26.5	23872720	0.08599	23696220	14000	7	1
27.0	24171760	0.09729	23942950	14000	7	3
27.5	24469020	0.10826	24182230	14000	7	5
28.0	24764390	0.11890	24414310	14000	7	7
28.5	25058030	0.12922	24639610	14000	7	9
29.0	25349930	0.13925	24858390	14000	7	11
29.5	25640230	0.14899	25071040	14000	7	13
30.0	25928800	0.15846	25277700	14000	7	15
30.5	26215850	0.16768	25479760	14000	7	17
31.0	26501290	0.17664	25674370	14000	7	19
31.5	26785230	0.18537	25864820	14000	7	21
32.0	27067630	0.19387	26050270	14000	7	23
32.5	27348550	0.20215	26230930	14000	7	25
33.0	27628030	0.21022	26407050	14000	7	27
33.5	27906140	0.21809	26578800	14000	7	29
34.0	28182810	0.22577	26746280	14000	7	31
34.5	28458230	0.23326	26909780	14000	7	33
35.0	28732240	0.24057	27069330	14000	7	35
35.5	29004970	0.24772	27225150	14000	7	37
36.0	29276430	0.25469	27377350	14000	7	39
36.5	29546510	0.26150	27526010	14000	7	41
37.0	29815540	0.26817	27671400	14000	7	43
37.5	30083250	0.27468	27913500	14000	7	45
38.0	30349860	0.28105	27952540	14000	7	47
38.5	30615190	0.28729	28058490	14000	7	49
39.0	30879380	0.29338	28221540	14000	7	51
39.5	31142550	0.29935	28351940	14000	7	53
40.0	31404510	0.30520	28479360	14000	7	55
40.5	31665420	0.31092	28604280	14000	7	57
41.0	31925240	0.31653	28726650	14000	7	59
41.5	32183960	0.32202	28846530	14000	8	1
42.0	32441730	0.32741	28964090	14000	8	3
42.5	32698450	0.33269	29079300	14000	8	5
43.0	32954160	0.33787	29192280	14000	8	7
43.5	33208930	0.34295	29303080	14000	8	9
44.0	33462600	0.34793	29411810	14000	8	11
44.5	33715410	0.35282	29518490	14000	8	13
45.0	33967210	0.35762	29623190	14000	8	15

The program for the transfer down

```
10 CLS
20 LET X=0
30 LPRINT " THETA          a (ft)              e              p (ft)        /\V/T (ft/s)   TOF
   (hr min)
40 LPRINT"---------------------------------------------------------------------------------
---"
50 U=1.4076E+16:PI=6.28:R2=1.782E+09:R1=2.182E+07
60 VC1=25399:VC2=10090:VPL=2595:DV1=6019:V1=77381
70 FOR TH=11 TO 191 STEP 10
80 T=(TH*(5400/360))+5400
90 AT=(((T/PI)^2)*U)^.333
100 E=1-(R1/AT)
110 P=AT*(1-E^2)
120 H=SQR(P*U)
130 VP=H/R1
140 DV2=V1-VP
150 DV3=VP-VC1
160 DVT=DV1+DV2+DV3
170 IF X=1 GOTO 220
180 TOF=T+18950
190 GOSUB 1000
200 LPRINT USING " ###          #######      #.#####      #######        #####
      ## ##";TH;AT;E;P;DVT;H;M
210 IF X=0 GOTO 230
220 LPRINT USING " ###                        #####                       ####
####";TH,DV1,DV2,DVT
230 NEXT TH
240 IF X=1 GOTO 300
250 LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT
:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LPRINT:LP
RINT
260 LPRINT" THETA             \V1 (ft/s)          \V2 (ft/s)        \V3 (ft/s)
270 LPRINT"---------------------------------------------------------------------------------
...
280 X=1
290 GOTO 50
300 END
1000 H=FIX(TOF/3600)
1005 M=((TOF/3600)-H)*60
1010 RETURN
```

Results of the program

THETA	a (ft)	e	p (ft)	/\VT (ft/s)	TOF (hr min)	
11	21901890	0.00374	21901590	14000	6	49
21	22293360	0.02123	22283310	14000	6	51
31	22681350	0.03798	22648640	14000	6	54
41	23066040	0.05402	22998730	14000	6	56
51	23447540	0.06941	23334570	14000	6	59
61	23825990	0.08419	23657100	14000	7	1
71	24201440	0.09840	23967100	14000	7	4
81	24573960	0.11207	24265330	14000	7	6
91	24943670	0.12523	24552500	14000	7	9
101	25310680	0.13791	24829270	14000	7	11
111	25675000	0.15015	25096190	14000	7	14
121	26036800	0.16196	25353860	14000	7	16
131	26396080	0.17336	25602760	14000	7	19
141	26752960	0.18439	25843370	14000	7	21
151	27107750	0.19505	26076040	14000	7	24
161	27459500	0.20538	26301290	14000	7	26
171	27809430	0.21537	26519460	14000	7	29
181	28157110	0.22506	26730970	14000	7	31
191	28502700	0.23446	26935880	14000	7	34
201	28846190	0.24357	27134790	14000	7	36
211	29187610	0.25242	27327860	14000	7	39
221	29527070	0.26102	27515370	14000	7	41
231	29864620	0.26937	27697650	14000	7	44
241	30200170	0.27749	27874780	14000	7	46
251	30533950	0.28539	28047110	14000	7	49
261	30865790	0.29307	28214750	14000	7	51
271	31195990	0.30055	28378030	14000	7	54
281	31524440	0.30784	28537040	14000	7	56
291	31851040	0.31494	28691910	14000	7	59
301	32175950	0.32186	28842890	14000	8	1
311	32499410	0.32860	28990130	14000	8	4
321	32821040	0.33519	29133770	14000	8	6
331	33141460	0.34161	29273930	14000	8	7
341	33460120	0.34789	29410750	14000	8	1
351	33777240	0.35400	29544340	14000	8	14
361	34092910	0.35999	29674860	14000	8	16
371	34407070	0.36583	29802370	14000	8	17
381	34719890	0.37154	29927050	14000	8	20

VII. PROPULSION

A high thrust nuclear engine was chosen for the propulsion system on the MOVERS orbital transfer vehicle.

This section will briefly review the history of high thrust nuclear engine development, and will outline the basic components of the engine used on the MOVERS OTV. The advantages of using these engines over more traditional chemical engines will then be described. The section will be concluded with a description of the potential problems with this engine. Included in that discussion will be an assessment of the environmental impact of using these engines.

History of High Thrust Nuclear Propulsion

A nuclear rocket propulsion system is not as exotic as it might first appear. Because it employs a light weight propellant (hydrogen) and can operate at very high temperatures, it is very efficient in its use of propellant. In fact, with existing technology, a nuclear engine can easily reduce propellant needs to less than one half that of a more traditional chemical rocket.

Given this potential, considerable research was performed by Los Alamos Laboratories during the 1960's and early 1970's under the Nuclear Engine for Rocket Vehicle Application (NERVA) program. The achievements of the NERVA program were significant: the final prototype which was built was started up 28 times and operated for a total of four hours with millions of gallons of hydrogen pumped through the engine. The engine had a reliability rating of .998, and was ready for actual flight testing in

space. This flight testing never occurred, though, because of the drastic cutbacks in space expenditures which occurred during the 1970's.

How the NERVA Engine Works

The basic principles behind the operation of a NERVA engine are elegantly simple. Essentially, a reactor is used to generate phenomenal amounts of heat energy. This energy is then picked up by the hydrogen propellant which is pumped directly through the reactor. The propellant is then expanded out through a nozzle, as is done with a chemical rocket, whereby it imparts momentum to the spacecraft.

Below is a schematic which details the basic components of these engines:
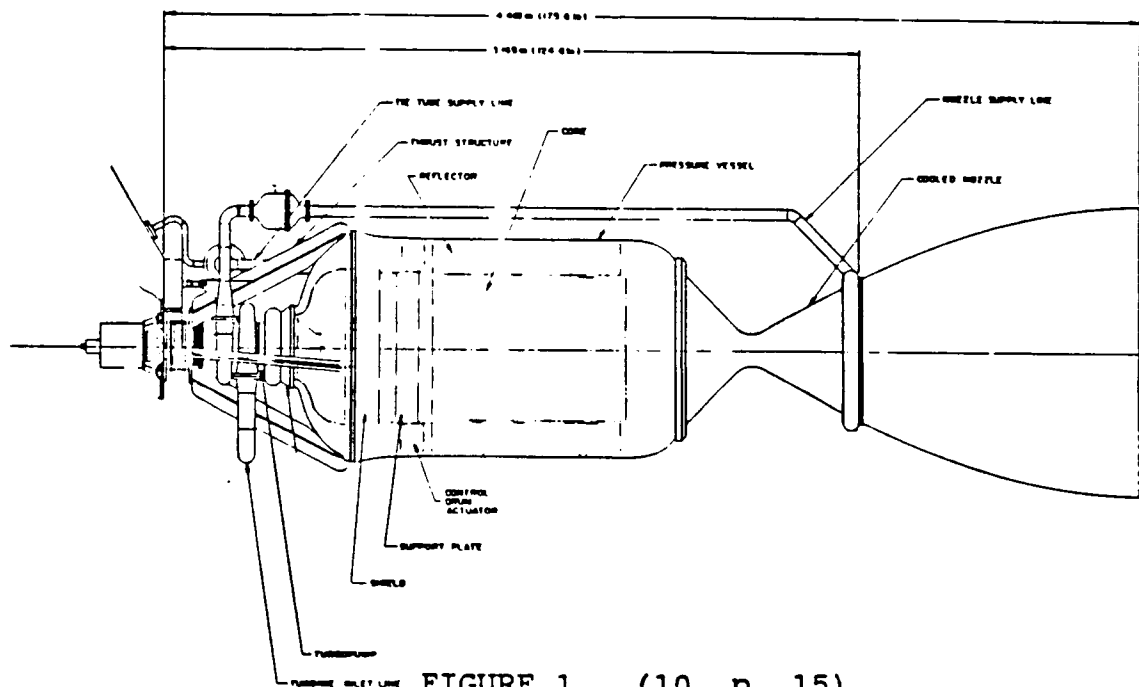
FIGURE 1.   (10, p. 15)

A High Thrust Nuclear Engine

The heart of the system is the reactor or solid fueled core. Here uranium (highly enriched U-235) is embedded in graphite rods. One end of each of these rods is attached to the core support grid; and the other end is left free so that the rod can expand as heat builds up in the core.

It should be noted that U235 is the preferred fuel for space reactors because its long half life insures that there will be no damage to the environment if the chemical rocket which carries the reactor into orbit, experiences a catastrophic failure.

To appreciate how the reactor is controlled, it is important to understand how nuclear fission works. a uranium atom fissions after it absorbs a neutron. When it fissions, it breaks into smaller fragments and releases more neutrons and considerable energy--which is what is used to heat the propellant. To control the fission process, there must be a strict balance between the number of neutrons which are released in fission, and the number which are available for absorption. The number of neutrons which ar released as a result of the fission is fixed by nature. Considerable control, however, can be exerted over the neutrons which are available for absorption.

The neutron reflector which surrounds the fuel elements, for example, is used to minimize the loss of neutrons out to space. If it were not there, more uranium atoms would be necessary to achieve the critical balance between neutrons absorbed and neutrons emitted. As such, these reflectors

decrease the amount of uranium which is required to achieve criticality--thereby reducing the overall weight of the system.

The fission control drums are the means by which the reactor is actually controlled. By rotating the drum, varying amounts of a neutron absorber (Boron-10) are exposed to the core. To start up the reactor, the drum is rotated to a position where no boron is exposed. The reactor goes supercritical (i.e., the number of neutrons available for absorption is greater than the amount currently being absorbed); and when the desired power level is achieved, the drums are adjusted for criticality (i.e., number of neutrons available for absorption equals number of neutrons being absorbed). To shut the reactor down, the drums are rotated to a position where a large amount of boron is exposed to the core--thereby making the core go subcritical (i.e., the number of neutrons available for absorption is less than the amount currently being absorbed). In this position, the fission process will soon stop.

With a NERVA reactor, the only limit to the amount of heat which can be generated is the melting point of the materials in the core. For the MOVERS OTV, the core/chamber will be designed to operate at a temperature of 4853 degrees Rankine (max temp 5256 degrees R), a pressure of 449.6 pounds/in^2, and a power density of 118.9 MWT/ft^3. This power density constitutes an order of magnitude increase over the power densities achievable with a chemical rocket. The engine will be capable of producing

30,000 pounds of thrust and will have a specific impulse of 880 seconds.

The nuclear shielding is a crucial part of the reactor design. While the reactor is operating, high energy photons are also being released as a natural result of the fission process. These photons would be dangerous to crews and equipment onboard both the OTV and nearby spacecraft. In addition, if the radiation is allowed to impinge upon the fuel tanks, it could cause the liquid hydrogen to enter the gaseous phase, which is considerably more difficult to handle. To limit this flux of radiation, a tungsten, lithium hydride shield was chosen for the OTV. What this shield does though, is to create a cone of protection as is depicted in the following diagram:
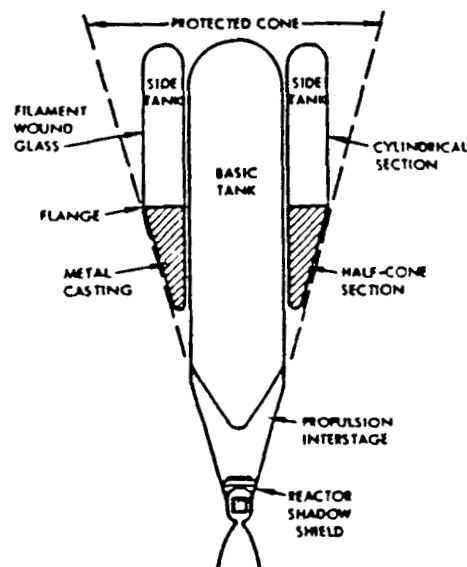
FIGURE 2. (9, p. 321)

Cone of Protection

People or equipment within the cone of protection would be protected against this high energy radiation. The shield is currently estimated to weigh 8500 pounds (Note Appendix C).

The use of a reaction control system to maneuver in close to target satellites may help to alleviate the problem of gamma radiation exposure; however, this would need to be further investigated.

The nozzle of the NERVA engine is patterned after standard chemical engines. The actual NERVA prototype was capable of sustaining temperatures from 36.6 R (20 K) up to 4500 R (2500 K), and pressures up to 90 atmospheres.

A crucial element of the nuclear engine, which was not depicted in the diagram above, is the turbopump. The turbopump's primary responsibility is to deliver hydrogen to the reactor. Rather than delivering the hydrogen directly to the chamber, though, the turbopump routes the gas around the chamber and rocket nozzle; in so doing, the turbopump not only provides cooling to the chamber and nozzle, but it also captures heat energy which would otherwise be lost--thereby improving the efficiency of the engine.

Reasons for Choosing a Nuclear Engine over a Chemical Engine

In selecting a propulsion system, the MOVERS design team considered the following propulsion/aerobrake systems:

1) Chemical engine
2) Chemical engine with aerobraking
3) Nuclear engine
4) Nuclear engine with aerobraking

In performing this analysis, two mission scenarios involving a spacecraft such as the MOVERS OTV (where the basic spacecraft less engine and tanks weighs 37,674 pounds) were considered. For the first scenario, the spacecraft was assumed to carry a 15,000 pound payload on a roundtrip from Low Earth (LEO) to Geosynchronous orbit (GEO). In the second scenario, an 80,000 pound payload was assumed to be carried on the same roundtrip. The assumptions and calculations associated with this study are presented in Appendix A.

The use of a nuclear engine in conjunction with an aerobrake resulted in the greatest savings. However, this option was not seriously considered because it was felt that the possibility of a catastrophic failure during the low-altitude, aerobrake pass posed too great of a risk to Earth's biosphere.

Of the remaining propulsion systems, it was found that a nuclear engine still used significantly less propellant than a spacecraft using a chemical engine, or a chemical engine/aerobrake system.

The propellant requirements for the 15,000 pound payload option for these propulsion/aerobrake options are presented in the following table:

TABLE 1

15,000 Pound Payload Option

Propulsion System	Propellant Required (lbs)	% Greater than Nuclear Engine Propel Requirements
Chemical	357,000	195.0%
Chem/Aerobrake	191,000	57.9%
Nuclear	121,000	

93

The dollar savings associated with the use of a nuclear engine for just one mission are staggering. Assuming that it costs $2000.00 to deliver a pound of propellant to the space station, the nuclear option results in a savings of $140.0 million dollars over the chemical/aerobrake option, and $472.0 million dollars over the chemical option.

The chemical/aerobrake system, which is somewhat competitive with the nuclear engine, does have a number of attractive qualities. First, it weighs significantly less than a nuclear engine (the combined weight of the chemical engine, aerobrake, oxygen and hydrogen tanks was 8300 pounds; whereas the weight of the nuclear engine, with its hydrogen tanks, was 19,000 pounds). And second, the use of an aerobrake in the chemical/aerobrake option resulted in a delta V savings of 7,000 ft/sec. Although these are significant attributes, it needs to be emphasized that they were not enough to offset the higher specific impulse (880 sec vs 460 sec) the nuclear engine.

In fact, the effect of the nuclear engine's higher Isp becomes more pronounced as the total mass of the spacecraft/ payload increases. For these larger vehicles, the weight penalty associated with the nuclear engine is simply a less significant factor in determining how much propellant is required to accomplish a given mission. This relationship is evident in the following table which outlines the propellant requirements for the 80,000 pound payload option:

94

C-2

TABLE 2

80,000 Pound Payload Option

Propulsion System	Propellant Required (lbs)	% Greater than Nuclear Engine Propel Requirements
Chemical	790,000	226.4%
Chem/Aerobrake	414,000	71.1%
Nuclear	242,000	

As can be seen by comparing the results of the 80,000 pound payload scenario to that of the 15,000 pound payload scenario, the chemical/aerobrake system requires proportionately more propellant than the nuclear engine in the 80,000 pound scenario than it does in the 15,000 pound scenario. Or stated another way, the nuclear engine does become increasingly more propellant efficient than the chemical/aerobrake system, as the total mass of the spacecraft/payload increases.

Thus the choice of a nuclear engine for the MOVERS OTV was based on two considerations. First, a nuclear engine uses significantly less propellant than either of the chemical systems studied. And second, a nuclear engine becomes even more propellant efficient as the overall mass of the spacecraft or payload increases--a vitally important consideration given that the trend in spacecraft design is towards heavier vehicles and payloads.

Potential Problems with Nuclear Engines

There are three environmental problems associated with the use of nuclear engines in space. First, there is a risk of gamma radiation exposure for crews and equipment on board the nuclear spacecraft as well as nearby. Second, there is also a risk of exposure for humans on the Earth as a result of nuclear spacecraft operations in Low Earth orbits. And finally, there is the very real issue of what is to be done with the spent nuclear reactors.

The issue of gamma radiation exposure for nearby spacecraft is not considered to be a serious problem. Spacecraft approaching the MOVERS OTV must exercise some caution. At long distances, they would be protected by the distance squared variation in the flux of gamma radiation. At short distances, though, they would have to be careful to approach within the cone of protection. Although considerable research is required to define these distances, this environmental problem does not appear to preclude the use of a nuclear engine.

The buildup of highly toxic, radioactive waste in the reactor, though, is a problem. It should be emphasized that this is only a problem when the nuclear spacecraft is operating in Low Earth orbits. Assuming that this is the case, there are essentially three ways that radioactive waste enter the Earth's biosphere. First, there can be leakage of fission byproducts through the fuel rods. Second, the nuclear spacecraft could experience a catastrophic failure in low Earth orbit. And

finally, if a burn is not aimed correctly, a nuclear spacecraft could follow an orbit which takes it right into the Earth's atmosphere.

The problem of leakage will be dealt with first. Although, the rods are designed to retain virtually all of the fission byproducts, some of this material will inevitably diffuse out through the rods into the hydrogen stream. This problem is more acute near the end of the engine's design life.

Analysis has shown that this problem is not significant. Using the program described in Appendix B, a very conservative estimate was made of the radiation exposure to humans on the surface of the Earth as the result of operating a NERVA engine in a 120 mile orbit. The engine was assumed to operate at 350 megawatts (thermal) for 10 consecutive hours in a 120 mile orbit. This scenario is absurd, as engines are used to go places and thus one half the burns would be made elsewhere. However, it will suffice to make a point. It was also assumed that 1% of the core material diffused out into the hydrogen propellant stream. With these assumptions, the radiation exposure to humans on the Earth was estimated to be 9.08 E-03 millirems. This radiation exposure level is less than 1% of the radiation damage which the average person receives from watching the television each year.

The issue of a catastrophic failure represents a more serious problem for nuclear spacecraft operations in Low Earth orbit. Accidents do occur, and the radiological impact of such

accidents must be assessed. Assuming a nuclear ship is rendered useless in a Low Earth orbit, the capability must exist either to destroy the reactor completely or to get it to a higher orbit where it will not soon re-enter the atmosphere. If the reactor re-enters the atmosphere intact, there is a distinct probability that it will survive intact all the way to the surface; in which case, the radiation exposures in the vicinity of the accident could be exceedingly dangerous. The crucial objective is to have the reactor break up at high altitudes so that its contents will be spread over as wide an area as possible.

The computer program described in Appendix B was used to assess the radiation damage to humans on the surface of the Earth, resulting from the catastrophic failure of a MOVERS OTV in a 120 miles orbit. The failure was assumed to occur at the end of the nuclear engine's design life, when the inventory of radioactive waste is greatest. The design life of the NERVA engine was assumed to be 10 hours.

The results of that analysis indicated that the exposure due to the NERVA engine was 0.91 millirems, which is well below the maximum allowable exposure of 500 millirems for the general population. Because a very conservative model was used, these results could easily exceed the correct values by an order of magnitude. Additional research is recommended to obtain more accurate results.

The last issue of concern regarding the buildup of radio-active waste in nuclear engines is that of what is to be done with the spent nuclear reactors. A number of recent studies on nuclear electric propulsion suggest that spent reactors be boosted to a 470 mile orbit. This orbit is referred to as a 300 year orbit because an object in this orbit will remain aloft for 300 years before it finally re-enters the Earth's atmosphere. It is argued that 300 years is sufficient to allow most of "the fission and activation products to decay before a reactor re-enters the atmosphere." (David Buden, "Space Reactors--What is a Kilogram"). In fact, the Russians are already storing their spent radioisotope therma generators in the 470 mile orbit.

There are several problems with this strategy. First, although the radiation exposure may not be significant if one reactor re-enters the Earth's atmosphere, the radiation exposure due to a large number of reactors re-entering the atmosphere could be significant. An even more pressing criticism is the fact, that storing spent reactor in this orbit only adds very heavy and very toxic pieces of junk to the growing and dangerous amount of space junk already floating around the Earth.

A number of possible alternatives exist for getting rid of spent reactors. They could be buried on the moon. They could be strapped together and sent into an orbit closer to the sun. And finally, reprocessing of the fuel in orbit may be possible. No conclusions are presented here; however, this problem needs to be addressed.

BIBLIOGRAPHY

01. Altseimer, J. H., et al., "The Nuclear Rocket Energy Center Concept," AIAA (AIAA paper AIAA-72-1091), 1972.

02. Angelo, J. P., "Shielding Considerations for Advanced Space Nuclear Reactor Systems," 1982 IEEE Conference on Nuclear and Space Radiation Effects, Las Vegas, Nevada, 1982.

03. Benedict, Manson, et al., Nuclear Chemical Engineering, McGraw Hill, New York, 1981.

04. Booth, L. A., et al., "Summary of Nuclear Engine Dual Mode Electrical Power System--Preliminary Study," Los Alamos Laboratories (Report LA-DC-72-1111), Los Alamos, N.M., 1972.

05. Buden, D., NERVA Systems Analysis Summary Report, Three Volumes, Aerojet General (Report #5001-CPO90290-F1), Aerojet, 1971.

06. Bureau of Radiological Health and the Training Institute Environmental Control Administration, Radiological Health Handbook, United States Government Printing Office, 1970.

07. Bussard, R. W., Nuclear Rocket Propulsion, McGraw Hill, New York, 1958.

08. Chamberlain, J. W. Theory of Planetary Atmospheres, Academic Press, New York, 1978.

09. Crouch, Holmes F., Nuclear Space Propulsion, Astronuclear Press, Grenada Hill, Ca., 1965.

10. Durham, Franklin P., Nuclear Engine Definition Study, (Vol 1-3) Los Alamos Laboratories, Los Alamos, New Mexico, 1972.

11. Goody, Richard M. Atmospheres, Prentice Hall, Englewood Cliffs, New Jersey, 1972.

12. Haffner, J. W., Radiation and Shielding in Space, Academic Press, New York, 1967.

13. Hill, Philip G., Mechanics and Thermodynamics of Propulsion, Addison-Wesley, REading, Massachusetts, 1970.

14. Holman, R. R., "A Critique of NERVA Based Nuclear Reactor Design and Performance for MMW Space Propulsion Applications," Space Nuclear Power Systems Conference, Albuquerque, N.M., 1987.

15. ---, "Development of NERVA Reactor for Space Nuclear Propulsion," AIAA 22nd Propulsion Conference, Huntsville, Ala., 1986.

16. Koenig, D. R., "Experience Gained from the Space Nuclear Rocket Program (ROVER),"

17. LaMarsh, John R., <u>Nuclear Engineering</u>, Addison-Wesley, Reading, Mass, 1983.

18. Morgan, K. Z., et. al., <u>Principles of Radiation Protection</u>, Robert E. Krieger Publishing Co., Malabar, Florida.

19. Technical Summary Report of NERVA Program Phase I: NRX and XN NERVA Fuel Development, Westinghouse Astronuclear Laboratory Report No. TNR-230, Westinghouse, 1972.

Appendix A

Notes RE: Propulsion Systems

To obtain a rough comparison of chemical vs. nuclear propulsion
systems, four options were studied:

1) Chemical
2) Chemical with Aerobrake
3) Nuclear
4) Nuclear with Aerobrake

The following assumptions were made:

Basic weights (in lbs):

Weight of the spacecraft (less: tank, engines, payload): 37,674

Weight of engines:
       Chemical (Based on RL 10 Engine) :   1,000
       Nuclear (Based on NERVA prototype): 12,500

Weight of Dry Tanks:

$$M (T) = pf * M (p)$$

where   M (T) = mass of tank
          pf = propellant fraction
            = ratio of tank mass to propellant mass

For Chemicals: pf = 0.0277 lbs tank/lb propellant *

For the NUC's: pf = 0.0545 lbs tank/lb propellant

Weight of payload: Scenario 1: 15,000
                     Scenario 2: 80,000

Weight of Aerobrake Shield: 2000 *

* Indicates value was obtained from Project Orion OTV study;
   Department of Aerospace Engineering, UVa, May 1988

To determine propellant required for each option, the following
relation was used:

$$M (O) / M (S) = e \char94 (delta \ V/I (sp) \ G)$$

Where M (o) = Total Mass of Vehicle before departure from
               low Earth orbit

$$= M \text{ (craft)} + M \text{ (tank)} + M \text{ (engine)}$$
$$+ M \text{ (propellant)}$$

$$= M \text{ (c)} + M \text{ (t)} + M \text{ (e)} + M \text{ (p)}$$

Where M (s) $= M$ (craft) $+ M$ (tank) $+ M$ (engine)

$$= M \text{ (c)} + M \text{ (t)} + M \text{ (e)}$$

NOTES: a) M (craft) is assumed to include the payload
b) for craft w/ aerobrake, M (e) also includes
the mass of the aerobrake

Where I (sp) = specific impulse

The following I (sp)'s were assumed:

Chemicals: 460 seconds

Nuclear: 850 seconds

The following delta V's were estimated for the three options:

01) Chemical: 28,000 ft/sec
02) Chemical w/ Aerobrake: 21,000 ft/sec
03) Nuclear: 28,000 ft/sec
04) Nuclear w/ Aerobrake: 21,000 ft/sec

Using this data, Mass ratio's were computed as follows:

01) Chemical: M(0) / M(s) = 6.62
02) Chem w/AB   M(0) / M(s) = 4.12
03) Nuclear: M(0) / M(s) = 2.78
04) Nuc w/AB   M(0) / M(s) = 2.15

The propellant requirements can then be determined according to:

$$M \text{ (p)} = MR \ (\ M \text{ (c)} + M \text{ (e)} \ / \ [\ 1 = (MR + PF \ MR) \ ] \ )$$

Using this relation, the propellant requirements were calculated
for each mission scenario:

Propellant Reqd (in lbs)

	15,000 lb Payld	80,000 lb Payld
01) Chemical:	357,000	790,000
02) Chem w/AB:	191,000	414,000
03) Nuclear:	121,000	242,000
04) Nuc w/AB:	76,200	152,000

Appendix B: Calculation of Radiation Exposure Due to
              Cataclysmic Failures of Nuclear Engines

Methodology Used

To calculate radiation exposures for humans on the Earth due to a cataclysmic failure of a nuclear engine in Low Earth orbits, it is necessary to determine the amount, and thus the activity, of radioactive particles in the nuclear engine at the time of failure, and then to determine the rate at which those particles work their way down through the atmosphere.   The total biological damage to humans at any given point in time is then a function of the concentrations of the various radio-nuclides in the air and the water.

The calculations associated with accurately determining the activities in a reactor and the rates of transport through the Earth's environment are exceedingly difficult.   As such, a considerably simpler model was used to calculate these quan-tities.   The model errs grossly on the side of conservatism, and the results for the exposure rates which were obtained could easily exceed the correct values by an order of magnitude. However, the results do provide a basic quantitative under-standing of the environmental issues associated with the use of nuclear engines in space.

To calculate the activity in the engine at the time of failure, the following model was used.   First, because it takes particles which are released at 120 miles approximately 2 to 5 years (18, p. 35) to work their way out of the stratosphere,

only those radioisotopes with significant half lives (T1/2> 50 days) were considered. The percentage of a given radioisotope in the reactor was then assumed to be equal to the cumulative percent fission yield by mass number. It should be noted that this significantly overstates the correct percentage because the cumulative fission yield also includes the contributions of short lived radionuclides of the same mass number.

With this simplification, the activity inside the reactor due to any given radionuclide is then given by:

$$N = Fy (1 - e^{-\lambda T_r}) e^{-\lambda T_c}$$

where: N = Atoms of long lived fission product after cooling for a time TC

λ = decay constant for the nuclide

F = Fission rate

y = Cumulative fission yield

Tr = irradiation time (i.e., time reactor is in opera-tion)

Tc = Cooling time (i.e., time material remains in orbit)

By expression F in terms of watts and the activity in terms of curies, and then dividing through by F yields the following:

$$N = Curies/watt = 0.845\, y\, (1 - e^{-\lambda T_R}) e^{-\lambda T_c}$$

Multiplying through by the total power level of the reactor, then yields the total activity due to a particular radionuclide

at any given time after the engine failure. The activities of each radionuclide are calculated in this manner.

It should be noted that a better way to calculate activities was found. Oak Ridge Laboratories devised a computer code known as ORIGEN which accounts for decay chains to provide accurate estimates of activities. Unfortunately, due to a lack of time, it was not possible to use this computer program. It would, nonetheless, be very interesting to employ this program in the environmental analysis of nuclear engine failures.

Once the activity of each radionuclide is calculated, it is then necessary to determine the biological damage resulting to humans on the Earth. A very simple, and conservative model was used. The radionuclide was assumed to remain suspended in the stratosphere for two years. After that time, these nuclides were assumed to be evenly distributed throughout the volume of air existing between the surface of the Earth and a height of two miles.

This is extremely conservative because it does not take into account the transport of particles through the atmosphere, the deposition of particles in the water bodies of the Earth, or geological processes such as erosion which bury and thus eliminate radionuclides from concern.

The assumption that the radionuclides remain suspended in air, and that the biological damage is due to inhalation is also conservative because the inhalation of radionuclides results

in orders of magnitude greater biological damage than the ingestion of radioactive particles.

To determine the actual radiation damage to humans on the Earth, the resulting concentrations of each radionuclide is divided by the maximum permissible concentration (MPC) of that nuclide to determine the radiation damage in rems. The total radiation damage is then the sum of the damage done by each radionuclide.

The attached computer program, RadExposure, is designed to perform all of the above calculations for radionuclides with sufficiently long half lives. Sample runs are also included.

Program RadExposure:

list 10-790

```
10    PRINT "This program is designed to calculate"
20    PRINT "radiation exposure to humans as a"
30    PRINT "result of a cataclysmic failure of a"
40    PRINT "nuclear engine"
45    PRINT
47    PRINT
50    REM   This program was designed by
60    REM   Richard McGuire Davis, on 03/25/88
70    PRINT "Select Type of Engine to be"
80    PRINT "investigated: "
85    PRINT
90    PRINT "NERVA Derivative: A"
100   PRINT "Nuclear Electric: B"
110   INPUT E$
120   PRINT
130   IF E$ = "B" THEN 200
135   IF E$ = "b" THEN 200
140   PRINT "Select Type of Engine Failure"
150   PRINT "to be Investigated: "
160   PRINT
170   PRINT "Engine Failure: A"
180   PRINT "Rod Leakage: B"
190   INPUT S$
200   REM   Assumptions employed:
210   REM   Fuel = U235;
220   REM   Reactor operated
230   REM   Continously;
240   REM   radioisotopes in
250   REM   orbits > 100 km have
260   REM   mean residence time
270   REM   in stratosphere =
280   REM   2 years;
500   REM   DATA ENTRY
510   REM   Following format is
520   REM   used:
530   REM   isotope, T1/2, time
540   REM   conversion, y(slow),
545   REM   y(fast), MPC(sol),
550   REM   MPC (insol)
700   REM   ACTUAL DATA
710   DATA   H3,12.4,y,1.3E-04,1.2E-04,3.0E-06,1.0E-07
720   DATA   Kr85,10.76,y,1.3,1.42,1,3E-06
723   DATA   Sr89,52.7,d,4.79,4.55,1E-04,1E-08
725   DATA   Sr90,27.7,y,5.77,5.59,3E-06,3E-10
730   DATA   Y91,58.8,d,5.84,5.41,3E-04,1E-08
740   DATA   Zr95,65.5,d,6.27,6.07,6E-04,1E-08
750   DATA   Ru106,368,d,.38,.901,1E-04,2E-09
760   DATA   Ag110,255,d,.02,.0757,3E-04,3E-09
770   DATA   Cd113,13.6,y,.0314,.0417,1E-07,1E-10
780   DATA   Sn119,250,d,.012,.0382,1E-07,1E-10
790   DATA   Sn123,125,d,.0173,.058,1E-07,1E-10
```

```
800   DATA   Sb125,2.71,y,.021,.0878,1E-03,9E-09
810   DATA   Cs134,2.046,y,8.06,7.25,4E-04,4E-09
820   DATA   Cs137,30.0,y,6.15,5.92,4E-04,5E-09
830   DATA   Ce144,284,d,5.62,5.83,1E-04,2E-09
840   DATA   Pm147,4.4,y,2.36,2.48,2E-03,2E-08
1000   REM   CALCULATIONS
1005   PRINT
1010   PRINT "Enter Nominal Power Level"
1020   INPUT "of Reactor (MWt-thermal): ";PL
1025   LET LEAKAGE = 100
1030   IF S$ = "B" OR S$ = "b" THEN   PRINT "Enter % of Core Material"
1040   IF S$ = "B" OR S$ = "b" THEN   PRINT "which leaks through the rods'

1050   IF S$ = "B" OR S$ = "b" THEN   INPUT "life of reactor: ";LEAKAGE
1055   REM   contamination zone
1060   PRINT "Enter Cooling period"
1062   INPUT "in orbit (yr): ";COOL
1065   REM   hcz = height of
1066   REM   contamination zone
1070   LET HCZ = 3220
1075   LET RE = 6.378E + 06
1080   DEF   FN A(X) = ((4 / 3) * 3.142 * (X ^ 3))
1090   LET VC =   FN A(RE + HCZ) -   FN A(RE)
1100   REM   Converting VC to CC
1110   LET VC = VC * 1.0E + 06
1120   PRINT "Enter design life of"
1125   INPUT "of engine (yr): ";TR
1200   REM   RAD EXPOSURE CALCS
1210   DIM N$(50),T12(50),TC$(50)
1220   DIM YS(50),YF(50),MS(50)
1230   DIM MI(50)
1235   DIM CW(50),TC(50)
1236   DIM CN(50),RI(50)
1240   FOR I = 1 TO 16
1250   READ N$(I),T12(I),TC$(I)
1260   READ YS(I),YF(I),MS(I)
1270   READ MI(I)
1280   IF E$ = "A" OR E$ = "a" THEN   LET Y = YS(I)
1290   IF E$ = "B" OR E$ = "b" THEN   LET Y = YF(I)
1300   LET A = 1 - ( EXP ( - 1 * TR * (.693 / T12(I)))))
1310   LET B =   EXP ( - 1 * COOL * (.693 / T12(I)))
1320   LET CW(I) = .845 * Y * A * B
1325   REM   CW(I) = curies of isotope/watt
1330   LET TC(I) = CW(I) * PL * 1E + 06 * (LEAKAGE / 100)
1335   REM   TC(i)=total curies of isotope
1336   REM   due to given power level
1340   LET CN(I) = (TC(I) * 1.0E + 06) / VC
1345   REM   CN(i) = microcurie concentration of isotope
1346   REM   for given volume of air
1350   LET RI(I) = (CN(I) / MI(I)) * 5
1355   REM   REM's due to concentration of isotope
1360   LET AC = AC + TC(I)
1365   REM   AC = Total Activity
```

```
1370   LET CT = CT + RI(I)
1375   REM  CT = Total REM exposure
1400   NEXT I
1490   PRINT "Isotope","Concen","REM Exp"
1500   FOR I = 1 TO 16
1510   PRINT N$(I),CN(I),RI(I)
1520   NEXT I
1525   PRINT
1550   IF S$ = "B" OR S$ = "b" THEN 1610
1560   PRINT "Total Activity = ";AC;" curies"
1610   PRINT "Total Exposure = ";CT;" rems"
2000   END
```

]

```
run

This program is designed to calculate
radiation exposure to humans as a
result of a cataclysmic failure of a
nuclear engine


Select Type of Engine to be
investigated:

NERVA Derivative: A
Nuclear Electric: B
?a

Select Type of Engine Failure
to be Investigated:

Engine Failure: A
Rod Leakage: B
?a

Enter Nominal Power Level
of Reactor (MWt-thermal): 350
Enter Cooling period
in orbit (yr): 2
Enter design life of
of engine (yr): .00114
Isotope                Concen                REM Exp
H3              1.32989429E-18          6.64947143E-11
Kr85             1.50669521E-14          2.51115868E-08
Sr89             1.25588321E-14          6.27941603E-06
Sr90             2.81068295E-14          4.68447159E-04
Y91             1.37608999E-14          6.88044994E-06
Zr95             1.32948688E-14          6.64743439E-06
Ru106            1.45928193E-16          3.64820483E-07
Ag110            1.10656931E-17          1.84428219E-08
Cd113            2.95781359E-16          1.4789068E-05
Sn119            6.77149175E-18          3.38574587E-07
Sn123            1.94165628E-17          9.70828141E-07
Sb125            6.59058282E-16          3.6614349E-07
Cs134            2.83790895E-13          3.54738618E-04
Cs137            2.77674169E-14          2.77674169E-05
Ce144            2.79353451E-15          6.98383627E-06
Pm147            5.55226345E-14          1.38806586E-05

Total Activity = 747441.173 curies
Total Exposure = 9.08498044E-04 rems

]
```

run

This program is designed to calculate
radiation exposure to humans as a
result of a cataclysmic failure of a
nuclear engine

Select Type of Engine to be
investigated:

NERVA Derivative: A
Nuclear Electric: B
?a

Select Type of Engine Failure
to be Investigated:

Engine Failure: A
Rod Leakage: B
?b

Enter Nominal Power Level
of Reactor (MWt-thermal): 350
Enter % of Core Material
which leaks through the rods
life of reactor: 1.0
Enter Cooling period
in orbit (yr): 2
Enter design life of
of engine (yr): .00114

Isotope	Concen	REM Exp
H3	1.32989429E-20	6.64947143E-13
Kr85	1.50669521E-16	2.51115868E-10
Sr89	1.25588321E-16	6.27941603E-08
Sr90	2.81068295E-16	4.68447159E-06
Y91	1.37608999E-16	6.88044994E-08
Zr95	1.32948688E-16	6.64743439E-08
Ru106	1.45928193E-18	3.64820483E-09
Ag110	1.10656931E-19	1.84428219E-10
Cd113	2.95781359E-18	1.4789068E-07
Sn119	6.77149175E-20	3.38574587E-09
Sn123	1.94165628E-19	9.70828142E-09
Sb125	6.59058282E-18	3.6614349E-09
Cs134	2.83790895E-15	3.54738618E-06
Cs137	2.77674169E-16	2.77674169E-07
Ce144	2.79353451E-17	6.98383627E-08
Pm147	5.55226345E-16	1.38806586E-07

Total Exposure = 9.08498045E-06 rems

]

Appendix C: Estimation of Radiation Shielding for MOVERS OTV

To accurately determine the radiation exposure which the crew of the MOVERS OTV would receive as a result of using the nuclear engine, it would be necessary to use a numerical technique such as the Monte Carlo method. Modeling such a problem, though, is very difficult; and was not possible to do within the time available.

Instead, an estimation of shielding requirements was obtained using a considerably simpler model which was found in Robert Busard's, Fundamentals of Nuclear Flight. In this model, the spacecraft is treated as consisting of the reactor, a shield, and a cylindrical hydrogen mass. The reactor is the radiation source, but its structure (i.e., the beryllium reflectors and aluminum pressure vessel) will absorb radiation. The shield consists of two separate parts: a tungsten shield to attenuate gamma photons and a lithium hydride shield to absorb neutrons. The hydrogen propellant, with its low atomic mass number, serves as an excellent absorber of neutrons.

The following steps are used to size the shield:

01) Determine the leakage of gamma radiation through the top of the reactor
02) Calculate the attenuation of that radiation through the top surface of the reactor and the hydrogen tank
03) Before the tungsten, gamma shield can be sized, the attenuation of the gamma radiation in the neutron, lithium-hydride shield must be known. Therefore, the leakage of (fast) neutrons through the top of the reactor must be estimated.
04) Next, the attenuation of the neutrons in the hydrogen tanks must be calculated. (Note: this model assumes minimal neutron attenuation in the reactor shell)

05) Once these calculations are performed, the size of the lithium-hydride shield can be estimated
06) The attenuation of gamma radiation in the LiH shield can then be determined. With this known, it will then be possible to size the tungsten, gamma shield.

The following pages outline these calculations.

Basic Information + Assumptions:

Reactor Power Level: 350 MWt

Operating or Burn time

For LEO-GEO-LEO $\Delta V = 28,000$ ft/sec

For the no payload option: Dry Mass: 56,775

$\frac{1}{2}$ Propellant (93,293): $\underline{46,674}$

Total: 103,422   (3212 slugs)

$M_{craft} \ a = thrust = 30,000$

$\therefore \quad a = \dfrac{30,000 \ lbs}{3212 \ slugs} = 9.34 \ ft/sec^2$

$\therefore$ To determine velocity: $at = \Delta V = 28,000 \ ft/sec$

$t = \dfrac{28,000 \ ft/sec}{9.34 \ ft/sec^2} = 3000 \ sec$

Partial fission Product Retention, $f_d = 0.80$

Reactor Core size (Assuming a square cylinder reactor) = 87.0 cm

Reactor Fuel: U235   # of fast neutrons produced per U235 fuel atom absorption, $\eta = 2.08$

Length of Propellant Tank: 36 ft $(1.10 \times 10^3 \ cm)$

Allowable total dose from engine per flight, from neutrons, $D_n^f = 0.1$ rads

from gammas, $D_\gamma = 1.0$ rads

The following Assumptions were Made:

01) Bulk core density, $\rho_{bc} = 1.3 \ gm/cm^3$

The core is made primarily of U235 and graphite + has a core void fraction, $f_{void} = 0.3$

02) Side + top beryllium reflectors have a thickness, $t_r = 12.0$ cm and a density, $\rho_r = 1.8 \ gm/cm^3$

03) The pressure vessel is made of Aluminum, has a thickness, $t_s = 3$ cm + a density, $\rho_s = 2.73 \ gm/cm^3$

115

Q4) The propellant, H_2 has a density $\rho_p = 0.071$ gm/cm³, Max Temp, $T_{MAX} = 5180\,R + \eta_J = 0.80$

01) <u>Calculation of Surface Leakage Flux Intensity, I_0</u>

The following values will be of help: Relevant Mass Absorption Coef: $\mu'_{Be} = 0.03$ cm²/

$\mu'_H = 0.07$ cm²/s

Carbon is assumed to be representative of the core composition $\mu'_{Al} = 0.035$ cm²/

3 MeV gamma photons are taken as an average $\mu'_c = 0.035$ cm²/s

The Surface Source Leakage Flux Intensity $I_0 = \frac{1}{2} \lambda_0 k_0$

where $\lambda_0 =$ relaxation Length $= \dfrac{1}{\mu'_c \, \rho_{ec}}$

$$= \frac{1}{\left(0.035 \frac{cm^2}{gm}\right)\left(1.3 \frac{gms}{cm^2}\right)} = 22.0 \text{ cm}$$

$k_0 =$ core power density $= \dfrac{10\left(1 + 0.8\, f_d\right) P_r}{181\left(1 + 0.08\, f_d\right) 2\pi \left(r_{core}\right)^3}$

$$= \frac{10\left[1 + 0.8\,(.8)\right] \; 350 \times 10^6 \text{ watts}}{181\left[1 + 0.08\,(.8)\right] \; 2\pi \left(\frac{57 \text{ cm}}{2}\right)^3}$$

$$= 57.6 \text{ watts/cm}^3$$

$$I_0 = \frac{1}{2}\left(22.0 \text{ cm}\right)\left(57.6 \text{ watts/cm}^3\right)$$
$$= 633.6 \text{ watts/cm}^2$$

02) <u>Determination of γ flux through the Be Reflector and Al chamber wall:</u>

In general, the attenuation of the flux is given by:

$$\frac{I_{P\gamma}(r)}{I_o} = E_1(\mu r) + e^{-\mu r}$$

where $E_1(\mu r)$ or in general $E_1(y)$ is the first exponential integral which is defined by: $E(x) = \int_x^\infty \frac{e^{-y}}{y} dy$

which for which are plotted in Appendix C [7, p.440]

$$\mu r = \mu'_{Be} \rho_r t_r + \mu'_{Al} \rho_s t_s$$

t_s is was noted = 3.0 cm

for the purposes of these calculations — the thickness is increased to simulate the γ attenuation which occurs in the walls of the crew compartment which also serve to protect the crew. Each wall is assumed to have a thickness of 2.2 cm + Thus our new $t_{s'} = 3.0$ cm + $2(2.2$ cm$)$
$$= 7.4 \text{ cm}$$

∴ the new $\mu r = \left[0.03 \frac{cm^2}{gm} \times 1.8 \frac{gm}{cm^3} \times 12.0 cm \right] + \left[0.035 \frac{cm^2}{gm} + 2.73 \frac{gm}{cm^3} \times 7.4 cm \right]$

$$= 1.355$$

Thus $\frac{I_{P\gamma}(r)}{I_o} = \left[E_1(1.355) + e^{-1.355} \right]$

$$I_{P\gamma}(r) = \left[0.20 + 0.26 \right] 633.6 \frac{watts}{cm^2}$$

$$= 291.4 \text{ watts}/cm^2 \quad (1.82 \times 10^{15} \text{ MeV}/sec \cdot cm^2)$$

03) Determination of γ attenuation in Hydrogen Tank:

In general, the integrated dose (over the burn time) @ the command module, D_γ is given by:

$$D_\gamma = \left[F_T \cdot D_{\gamma_{reac \; Top}} \cdot t_b \cdot \frac{1 hr}{3600 \, sec} \right] A_f \left[\frac{d_{reac \; Top}}{d_{crew \; sep}} \right]^2$$

where F_T = Fraction of Reactor Top Radiation which actually impinges on the tanks
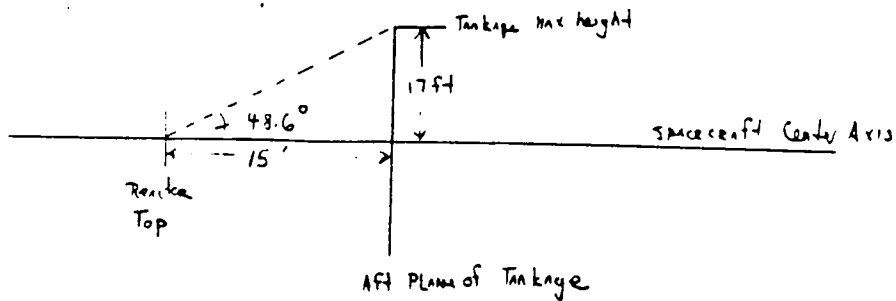
Assuming an isentropic distribution of radiation, then $\frac{48.6}{90} \cong 50\%$ of radiation from reactor top will impinge on tankage.

$$D_{\gamma_{reac\ Top}} = \text{Dose Rate @ Reactor top}$$

$$= I_{p_{\gamma_{reac\ Top}}} / 5.6 \times 10^5 \frac{MeV/sec \cdot cm^2}{Rad/hr}$$

$$= \frac{1.81 \times 10^{15}\ MeV/sec \cdot cm^2}{5.6 \times 10^5\ MeV/sec \cdot cm^2/rad/hr}$$

$$= 3.25 \times 10^9\ rad/hr.$$

A_t = Attenuation coefficient

$$= \left[2 - (2 + \mu_p h_p) \exp(-\mu_p h_p) \right] / \mu_p h_p$$

where μ_p = 3 MeV absorption coefficient for Hydrogen

$$= \mu_p' \rho_{H_2}$$

$$= 4.2 \times 10^{-3}\ cm^{-1}$$

$$\mu_p h_p = (4.2 \times 10^{-3}\ cm^{-1})(1.10 \times 10^3\ cm)$$

$$= 4.62$$

$$A_t = \left[2 - (2 + 4.62)\ \ell^{-4.62} \right] / 4.62$$

$$= 0.42$$

$d_{reac\ Top}$ = distance from reactor centr to top of reactor

$$= \frac{87}{2}\ cm\ +\ t_r\ +\ t_s$$

$$= [43.5 + 12.0 + 3.6]\ cm$$

$$= 58.5$$

$d_{crew\ sep}$ = distance from reactor centr to centr of command Module

$$= 105\ ft\ \times\ 12\ \frac{in}{ft}\ \times\ 2.54\ \frac{cm}{in}$$

$$= 3200\ cm$$

$$\therefore\ D_\gamma = \left[0.50 \cdot 3.25 \times 10^9\ \frac{rad}{hr} \cdot 3000\ sec \cdot \frac{1\ hr}{3600\ sec} \right] 0.42 \left[\frac{58.5\ cm}{3200\ cm} \right]^2$$

$$= 1.90 \times 10^5\ rads.$$

If we postulate that $\frac{2}{3}$ of the γ dose is Received during powered operation + $\frac{1}{3}$ in the first 1.5 hrs following reactor shutdown, the total Dose will be:

$$D_{\gamma_T} = \frac{D_\gamma}{2/3} = 2.85 \times 10^5\ rads.$$

NOTE: This is the γ attenuation which must be achieved using Both the γ, tungsten + the LiH, neutron shield. Thus we must first determine the LiH shield + to do this we must know the fast neutron leakage from the reactor.

04) Determination of fast neutron leakage:

In general, the fast neutron leakage, $I_{En}^{ACT} = (1 - P_f)\ I_{En}^{MAX}$

where P_f = non-Leakage Probability
$$= e^{-B^2 \tau_{TH}}$$

where B^2 = Geometric Buckling Factor

$$= \left[\frac{2.405}{r_{bc}}\right]^2 + \left[\frac{\pi}{L_c}\right]^2$$

w/ r_{bc} = radius of bare core

$\qquad = \frac{1}{2}$ Diameter bare core, D_c

$\qquad = \frac{1}{2}\left[D_{Act\ core} + 2(reflector\ saving \cdot s)\right]$

w/ $s = \dfrac{D_c}{D_r}\ \ell_r$

where D_c = Core diffusion coefficient

$$= \frac{1}{3\,\Sigma_{s_c}(1 - 2/3A_c)}$$

A_c = Atom Mass - carbon

Σ_s = Scattering Cross Section

$$= \frac{1}{3(0.3811)(1 - 2/3 \cdot 12.01)}$$

$$= 0.926$$

$$D_r = \frac{1}{3\,\Sigma_{s_{Be}}(1 - 2/3 A_{Be})}$$

$$= \frac{1}{3(.7589)(1 - 2/3 \cdot 9.01)}$$

$$= 0.474$$

$\therefore\ \ s = \dfrac{0.926}{0.474}\ \ 12.0\ cm$

$$= 23.4\ cm$$

$\therefore\ r_{bc} = \frac{1}{2}\left[87\,cm + 2(23.4\ cm)\right]$

$$= 66.9\ cm$$

L_c = Bare Core Length

$\quad = D_{ec} + S$

$\quad = 87\ cm + 23.4\ cm$

$\quad = 110.4\ cm$

$$\therefore \quad B^2 = \left[\frac{2.405}{66.9\ cm}\right]^2 + \left[\frac{\pi}{110.4\ cm}\right]^2$$

$$= 2.10 \times 10^{-3}\ cm^{-2}$$

where $\tau_{TH'}$ = age to thermal in core (ie. graphite) accounting for core void

$$= \tau_{TH_c} / (1 - P_f)^2$$

$$= 344\ cm^2 / (1 - 0.3)^2$$

$$= 702\ cm^2$$

$$\therefore\ P_f = e^{(-2.10 \times 10^{-3}\ cm^{-2})(702\ cm^2)}$$

$$= 0.23$$

where $I_{en}^{MAX} = \dfrac{1.38 \times 10^{16} \left(1 - 1/\eta\right)\ P_r}{\left(1.0 + 0.08\ f_d\right)\ r_c^2} \qquad <(MeV/sec)/cm^2>$

$$= \frac{\left(1.38 \times 10^{16}\ MeV/sec\,/MW\right)\left(1 - 1/2.05\right)\left(350\ MW\right)}{\left[1.0 + 0.08\ (.F)\right]\left[66.9\ cm\right]^2}$$

$$= 5.27 \times 10^{14}\ (MeV/sec)/cm^2$$

Recalling that $I_{en}^{ACT} = \left(1 - P_f\right) I_{en}^{MAX}$

$$= \left(1 - 0.23\right) 5.27 \times 10^{14}\ (MeV/sec)/cm^2$$

$$= 4.06 \times 10^{14}\ (MeV/sec)/cm^2$$

In turn, the fast neutron dose just outside the pressure shell, assuming minimal attenuation is given by:

$$D_n^f (r_{reactor}) = \frac{I e^{-\mu r}}{2.60 \times 10^5 \; (MeV/sec)/cm^2} \; rads/hr.$$

$$= \frac{4.06 \times 10^{14} \; (MeV/sec)/cm^2}{2.60 \times 10^5 \; (MeV/sec)/cm^2} \; rads/hr.$$

$$= 1.56 \times 10^9 \; rads/hr.$$

05) <u>Determination of fast Neutron Attenuation in Hydrogen Tanks:</u>

In general, the net neutron dose due to reactor operation is given by:

$$D_n^f = \frac{D_n^f (r_{reac\,top}) \, t_b \cdot 1hr/3600\,sec}{\Sigma_r^f \, h_p} \left(1 - e^{-h_p \Sigma_r^f}\right) \left[\frac{r_{bc}}{d_{crew\,sep}}\right]^2$$

$$h_p \Sigma_n^f = \left(1.10 \times 10^3 \, cm\right)\left(0.04 \; cm^{-1}\right)$$
$$= 44$$

$$\therefore D_n^f = \frac{\left(1.56 \times 10^9 \; rad/hr\right)\left(3000\,sec\right)\left(1hr/3600\,sec\right)}{44} \left[\frac{66.9 \, cm}{3200 \, cm}\right]^2$$

$$= 12,913 \; rads$$

06) <u>Determination of mass of LiH, Neutron Shield:</u>

The thickness of the shield is given by:
$$X = \Sigma_r^f \, t_{LiH}$$
$$\text{or} \quad t_{LiH} = \frac{X}{\Sigma_r^f}$$

To determine x, we know that we want an attenuation, B_s given by:

$$B_s = \frac{D_n^f}{0.1 \text{ rad}/\text{Trip}} = \frac{12,913 \text{ rads}}{0.1}$$

$$= 129,130$$

$$E_1(x) = \frac{1}{B_s} = \frac{1}{1.29 \times 10^5} = 7.8 \times 10^{-6}$$

Upon refering to appendix C $[7, p.440]$, $x = 9.1$

Given that $\Sigma_r^f = 0.152 \text{ cm}^{-1}$ for LiH

$$t_{LiH} = \frac{(9.1)}{0.152 \text{ cm}^{-1} \text{ LiH}} = 60 \text{ cm LiH}$$

$\therefore$ Mass of LiH shield is given by:

$$M_{LiH} = t_{LiH} \, \rho_{LiH} \, \pi \left(1.2 \cdot d_{core \, Top} \right)^2$$

NOTE: assume shield extends 20% beyond width of core

$$= (6.0 \text{ cm})\left(0.82 \frac{gm}{cm^3} \right) \pi \left(1.2 \cdot 57.5 \text{ cm} \right)^2$$

$$= 7.62 \times 10^5 \text{ gms}$$
$$= 1670 \text{ lbs}$$

o 7) <u>Determination of γ attenuation in LiH shield</u>:

The attenuation of γ radiation in the LiH shield is given by $\frac{I_{P\gamma}(r)}{I_0} = E_1 (\mu r)_{LiH} + e^{-(\mu r)_{LiH}}$

where $(\mu r)_{LiH} = 60 \text{ cm LiH} / 36 \text{ cm}$
$$= 1.67$$

123

The attenuation of γ radiation is given by:

$$\frac{I_{P\gamma}(r)}{I_o} = E_1(1.67) + e^{-1.67}$$

$$= 0.20 + 0.19$$

$$= 0.39$$

Thus the additional gamma shielding must shield against:

$$D_{P\gamma_T}(r) = 0.39 \ D_{o_T}$$

$$= 0.39 \ (2.85 \times 10^5 \ \text{rds})$$

$$= 1.11 \times 10^5 \ \text{rds}.$$

08) Sizing of Tungsten, gamma shield:

We seek an attenuation $= \dfrac{1}{D_{\gamma T}} \simeq E_1(x) = E_1(\mu t_w)$

$$\therefore E_1(\mu t_w) = \frac{1}{1.11 \times 10^5 \ \text{rds}} = 9.00 \times 10^{-6}$$

from Appendix C $\left[7, \ p.440 \right]$

$$\mu_w \ t_w = 11.1$$

$$\rho_w \ t = 11.1 / \mu'_w$$

$$= 11.1 / 0.83 \ \text{cm}^{-1}$$

$$t_w = 13.4 \ \text{cm}.$$

$\therefore$ The mass of the shield is: $\rho_w \ t_w \pi \left(1.2 \cdot 58.5 \ \text{cm}\right)^2$

$$= \left(19.3 \ \text{gr/cm}^3\right)\left(11.6 \ \text{cm}\right) \pi \left(4930 \ \text{cm}^2\right)$$

$$= 3.47 \times 10^6 \ \text{gm}$$

$$= 7650 \ \text{lbs}.$$

09) Correcting LiH shield to account for attenuation of neutrons in tungsten, γ shield:

$$\text{The attenuation of neutrons} = \Sigma_{r_w}^{t} \, t_w$$

$$= 0.221 \text{ cm}^{-1} \cdot 11.1 \text{ cm}$$

$$= 2.45$$

Thus the actual attenuation of neutrons which is needed is: $\dfrac{12,913 \text{ rads}}{2.45} = 5270 \text{ rads}$

$$E_1 \left(\Sigma_{r_{LiH}} \, t_{LiH} \right) = \frac{1}{5270} = 1.90 \times 10^{-4}$$

referring to Appendix C [7, p. 440] $\quad \Sigma_{r_{LiH}} t_{LiH} = 6.7$

$$t_{LiH} = 6.7 / 0.152 \text{ cm}^{-1}$$

$$= 44.1 \text{ cm}$$

$$M_{LiH} = \rho_{LiH} \, t_{LiH} \, \pi \left(1.2 \cdot d_{reactor} \right)^2$$

$$= 0.82 \frac{gm}{cm^3} \cdot 44.1 \text{ cm} \cdot \pi \left(1.2 \cdot 58.5 \text{ cm} \right)^2$$

$$= 5.60 \times 10^5 \text{ gms}$$

$$= 1235 \text{ lbs}$$

Thus the total shield weight $= M_{LiH} + M_w$

$$= 1235 \text{ lb} + 7650 \text{ lbs}$$

$$= 8885 \text{ lbs}$$

Due to approximation used which err on the side of conservatism, the shield weight will be taken to be 8500 lbs.

VIII. REACTION CONTROL SYSTEM AND TANKAGE

The purpose of the reaction control system (RCS) is to control the orientation of the spacecraft by producing small rotational and translational velocities which are independent of the main propulsion system (MPS). The RCS must have a fail-safe / fail-operational design to insure man-rating requirements, but also have a reasonable development cost.

RCS Requirements

In addition to the above general requirements, a report by the Boeing Aerospace Co. listed other important and necessary RCS requirements [5,p.212].

1. Provide thrust for velocity maneuvers of less than 20 fps.
2. Satisfy man-rating requirements.
3. Be compatible with shuttle launch.
4. Have a 20 mission reuse capabilty.
5. Provide six degrees of freedom control for docking maneuvers.
6. Be capable of operating in a space-based mode (ground-basing of the OTV presented in this report is impractical).
7. Satisfy the following performance requirements.

Mission Phase	Delta-V (fps)
Separate from Station	10
Coast to first perigee burn	0
Intermediate orbit transfer	20
Transfer to GEO coast	10

GEO payload positioning	15
Coast at GEO (24 hours)	50
Deorbit to LEO coast	50
Phasing orbit trim	15
LEO circulation	10
Station rendezvous and docking	10
Total	190

RCS Candidates

Three types of reaction control systems were studied for the space-based, nuclear-propulsion OTV; they were: (1) monomethyl-hydrazine/nitrogen tetroxide, (2) supercritical hydrogen/oxygen, and (3) monopropellant hydrazine.

1. MONOMETHYL-HYDRAZINE / NITROGEN TETROXIDE

The monomethyl-hydrazine/nitrogen tetroxide RCS (MMH/NTO) is what the shuttle fleet presently uses, thus its technology is state-of-the-art. The most attractive feature of this system is that no ignition system is required; once the fuel and oxidizer come into contact, combustion occurs spontaneously. Although this feature adds reliability to the system and saves weight due to a simplified thruster design, the toxic and corrosive nature of the propellants can pose a serious risk during on-orbit refuelings. However, the propellants can remain in their liquid states for long periods of time without decomposing or boil-offing like cryogenic systems.

The fuel and oxidizer tanks are pressurized to 400 psia by gHe to insure a positive propellant flow into each thruster. Electric heaters around propellant lines are utilized to prevent freezing during low RCS activity. The specific impulse is 280 seconds with a mixture ratio of 1.65.

2. SUPERCRITICAL HYDROGEN / OXYGEN

The supercritical hydrogen/oxygen RCS is attractive for use on the OTV for a number of reasons. The propellants have common storage with the fuel cell reactants, thus minimizing the number of fluids that require refueling at the Space Station. The propellants are noncorrosive and relatively clean and nontoxic - all attributes desired in a reuseable system. The specific impulse is conservatively estimated at 410 seconds, and therefore, this RCS would have the lowest wet mass.

Some of the disadvantages are that high pressure, lightweight composite tanks for containing hydrogen at 300 psia and oxygen at 900 psia need to be thoroughly tested. Rocket thruster development and testing is requirted since hydrogen and oxygen have never been used for RCS maneuvers. These disadvantages all add up to a research and development cost which was estimated by the Martin Marietta Aerospace Co. in 1984 to be $166 million [7.p.5].

3. MONOPROPELLANT HYDRAZINE

The last RCS that was studied was the hydrazine system. It has been used successfully on the Gemini and Apollo Space Pro-

grams, and it is used extensively today on satellites and space probes. The hydrazine RCS has the lowest dry weight but requires more propellant due to its lower specific impulse of 230 seconds. The attractive features of this system are its proven reliabilty and simplicity - inherent to its single propellant design. As with the MMH/NTO RCS, electic heaters are required, but the mono-propellant system is pressurized by gN2 to 380 psia.

Comparison and Selection

The supercritical hydrogen/oxygen RCS is recommended for use on the nuclear-propulsion OTV because of its higher performance and lower total program cost. A subsystem trade study prepared by Boeing in 1986 showed that the hydrazine RCS had a slight advantage over the cryo system [5,p.219]. However, the nuclear-propulsion, non-aerobraked OTV required a larger amount of MPS propellant than the Boeing ballute braked OTV. Therefore, it was believed that the increased RCS thrust level needed to overcome the larger moments of inertia of the nuclear engine and MPS propellant offset the slight advantage of the hydrazine system by increasing the 385 lbm total weight advantage of the cryo system. The net resupply benefit per flight would be greater than 660 lbm making the hydrogen/oxygen system's total program cost lower than the hydrazine RCS. The trade study comparisons and concept of the cryo RCS are illustrated on the following pages.

Dual redundancy for manned space missions requires 24 RCS thrusters to satisfy six degree of freedom control. The thrust-

130

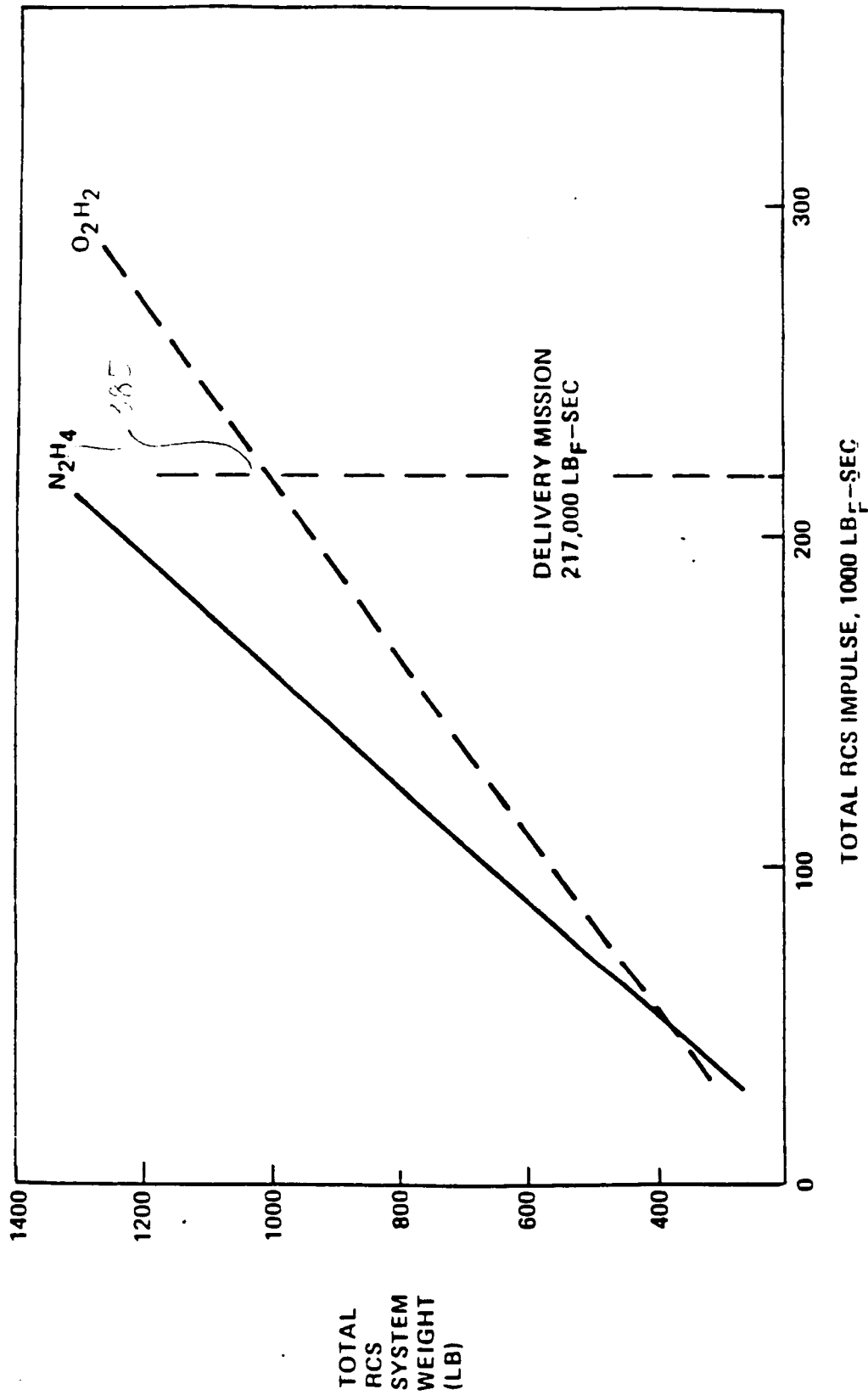ers are arranged in four clusters with six thrusters in each.

OTV 942

Figure 4.2.3 Reaction Control System Weight Comparison

Table 4.2-2 System Characteristics for RCS Concepts

OPTIONS

- N_2H_4
  - EXTERNAL PRESSURIZATION
  - $I_{SP} = 220$ SEC.

- O_2/H_2
  - SUPERCRITICAL STORAGE
    - O_2 AT 900 PSIA; H_2 AT 300 PSIA
  - COMMON WITH FUEL CELL STORAGE
  - $I_{SP} = 410$ SEC.

KEY GROUND RULES

- TOTAL ΔV BUDGET —185 FPS
- FAIL OP/FAIL SAFE
- 24 THRUSTERS
- 25 LB_F EACH

TYPICAL WEIGHT (LBM)

(GEO PAYLOAD 20 KLBS)

	N_2H_4	O_2/H_2
• DRY	280	355
• RCS PROP	990	530
• MPS PROP	66680	66480

NET RESUPPLY BENEFIT OF THE O_2/H_2 SYSTEM PER FLIGHT IS 660 LB

OTV 682

DISCOUNTED

3.15

3.14

N₂H₄   O₂H₂
PROPELLANT

5

4

3

2

1

0

BILLINGS
OF
DOLLARS

● SB B/B OTV
● LOW MODEL (1997–2010)

UNDISCOUNTED

17.4

17.4

N₂H₄   O₂H₂
PROPELLANT

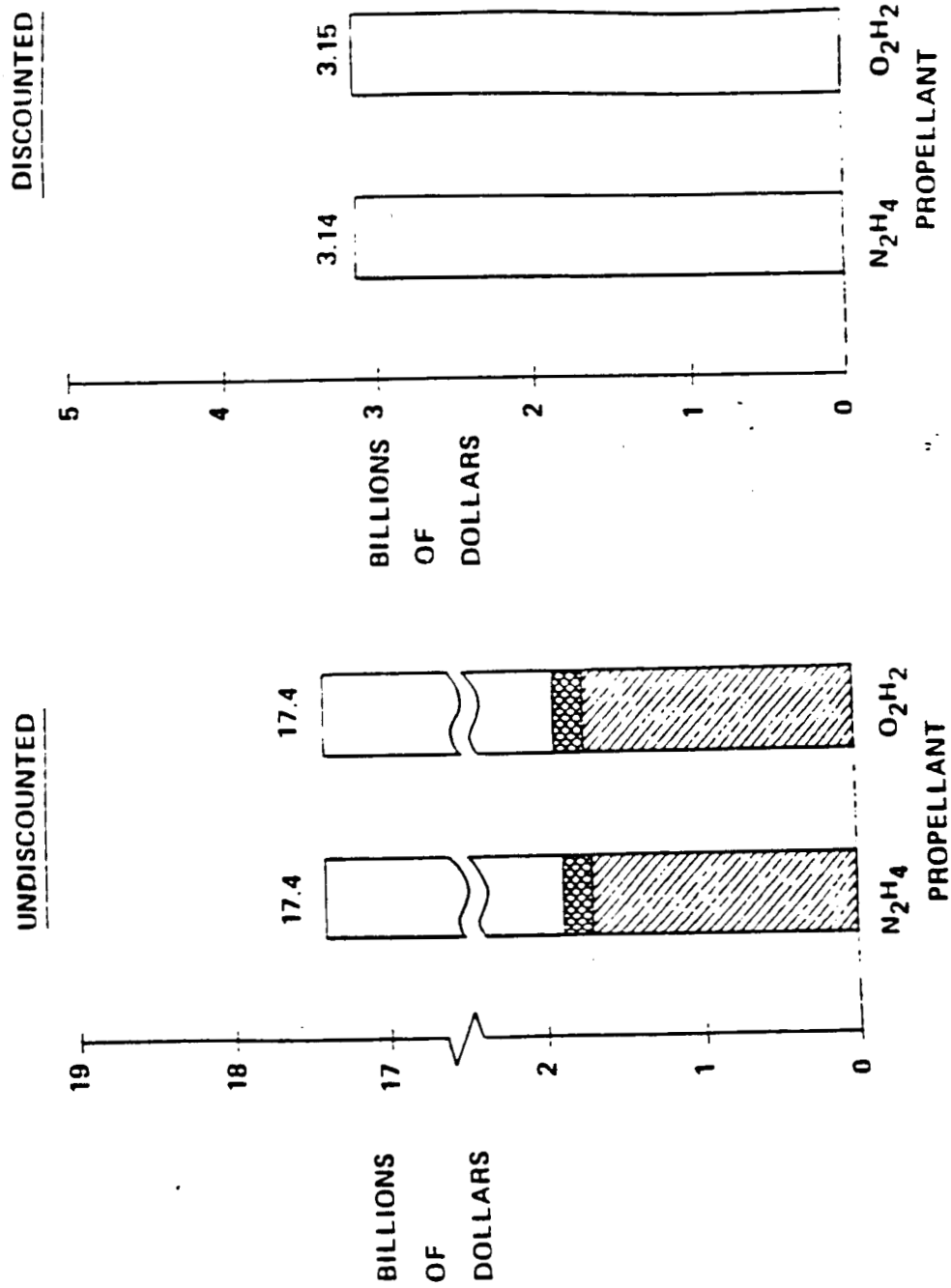19

18

17

2

1

0

BILLINGS
OF
DOLLARS

Figure 4.2-4 OTV Program LCC Comparison RCS Influence

OTV-1000

- SYSTEM CHARACTERISTICS
  - PROPELLANT STORAGE COMMON WITH FUELS CELLS
  - THRUST, 25 LBF
  - SPECIFIC IMPULSE 410 SEC. (CONSERVATIVE)

- ADVANTAGES
  - LOGISTICS-MINIMUM NO. OF FLUIDS REQUIRED
  - NON-CONTAMINATING EXHAUST
  - MINIMUM NO. COMPONENTS FOR RCS PLUS FUEL CELLS.

- DISADVANTAGES
  - ADVANCED TECHNOLOGY LIGHT WEIGHT TANKS REQUIRED
  - THRUSTER DEVELOPMENT REQUIRED. (BEING DONE FOR SPACE STATION.)

- CONCERNS
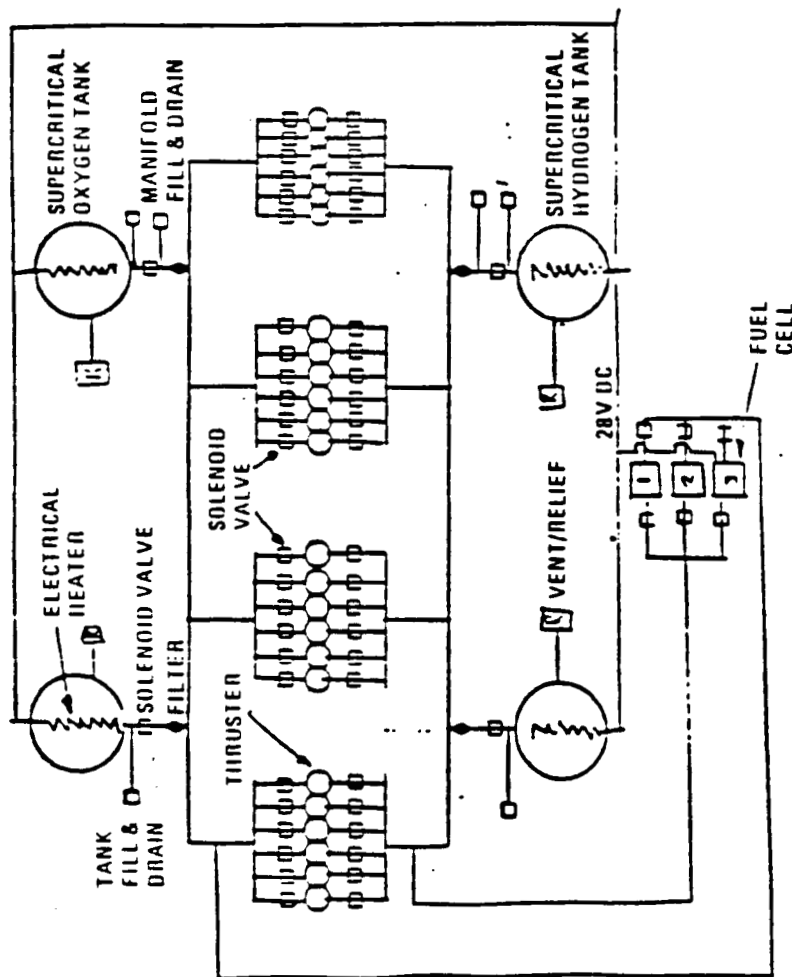 - TRANSIENT PRESSURES WITH CRYOGENIC FLUIDS AND WARM LINES.

Figure 4.2.2 Reaction Control Subsystem—Supercritical Oxygen-Hydrogen

135

ORIGINAL PAGE IS OF POOR QUALITY

MPS Tanks

The configurations of the MPS tanks are illustrated in the Overview section of this report. For the mission to GEO without a payload, four cylindrical tanks having a length of 39 ft and a diameter of 14 ft were selected because this tank design gave the best volume of propellant for packaging within the shuttle cargo bay. Each tank accommodates approximately 23,323 lbm of LH2 at 15 psia and is made of 2219-T62 Aluminum.

For the mission to GEO with a 15,000 lbm payload, an additional propellant module is required. This module is also illustrated in the Overview section, and it is comprised of four cylindrical tanks with lengths of 15 ft and diameters of 14 ft. The addiitonal tanks are made of the same material and can accommodate approximately 6,970 lbm of LH2 per tank.

The results stated above are summarized on the following pages, and the computer program used to determine these results is in the Appendix.

THE OTU PARAMETERS ARE AS FOLLOWS:

<1>   800   <5>   37,674

<2>   4.42   <6>   0

<3>   28,000   <7>   12,500

<4>   0.0545

RESULTS

148,549

93,293

62.80

21,107

POSSIBLE TANKAGE CONFIGURATIONS

21,107.0

14.0

14.0, 25.00

		15
SPHERICAL	1,436.76	
CYLINDRICAL	5,285.21	4
combination SPHERES & CYLINDERS	Average Vol: 5,285.21	Sph: 0 Cyl: 4

THE OTV PARAMETERS ARE AS FOLLOWS:

(1) 880

(2) 4.42

(3) 28,000

(4) 0.0545

(5) 37,674

(6) 15,000

(7) 12,500

RESULTS

192,959

121,184

62.80

27,417

POSSIBLE TANKAGE CONFIGURATIONS

27,417.1

14.0

14.0, 35.20

SPHERICAL	1,436.76	20
CYLINDRICAL	6,855.37	4
combination SPHERES & CYLINDERS	Average Vol: 6,855.37	Sph: 0 Cyl: 4

The LH2 tanks will be removable for space maintenance and replacement, and will have the technological concepts of the spherical removable cryo tank illustrated on the next page. The following concepts are incorporated in the design: a single fluid / electrical disconnect, a start trap to minimize the time spent in tank level idle during pump conditions, magnetic drive motors for the thermodynamic vent system (TVS), autogenous pressurization, and pre-chill spray nozzles for onorbit refill.

The thermal protection system (TPS) is composed of 50 layers of MLI having a density of approximately 3.5 lbm/cubic ft. This resulted in a blanket thickness of 1 inch per LH2 tank to minimize boil-off losses. A concern regarding this type of TPS is the possibility of delamination due to the high energy photons from the nuclear engine.

REMOVABLE CRYO TANK CONCEPT

VENT/RELIEF VALVE

PRE-CHILL SPRAY NOZZLES

POINT SENSORS

BAFFLES

START SUMP

SIPHON ASSY

DIFFUSER

PRESS. LINE

SPACE MULTI-FEED DISCONNECT

FEED LINE

TVS HEAT EXCH

MAGNETICALLY COUPLED PUMP AND MOTOR

MARTIN MARIETTA

144
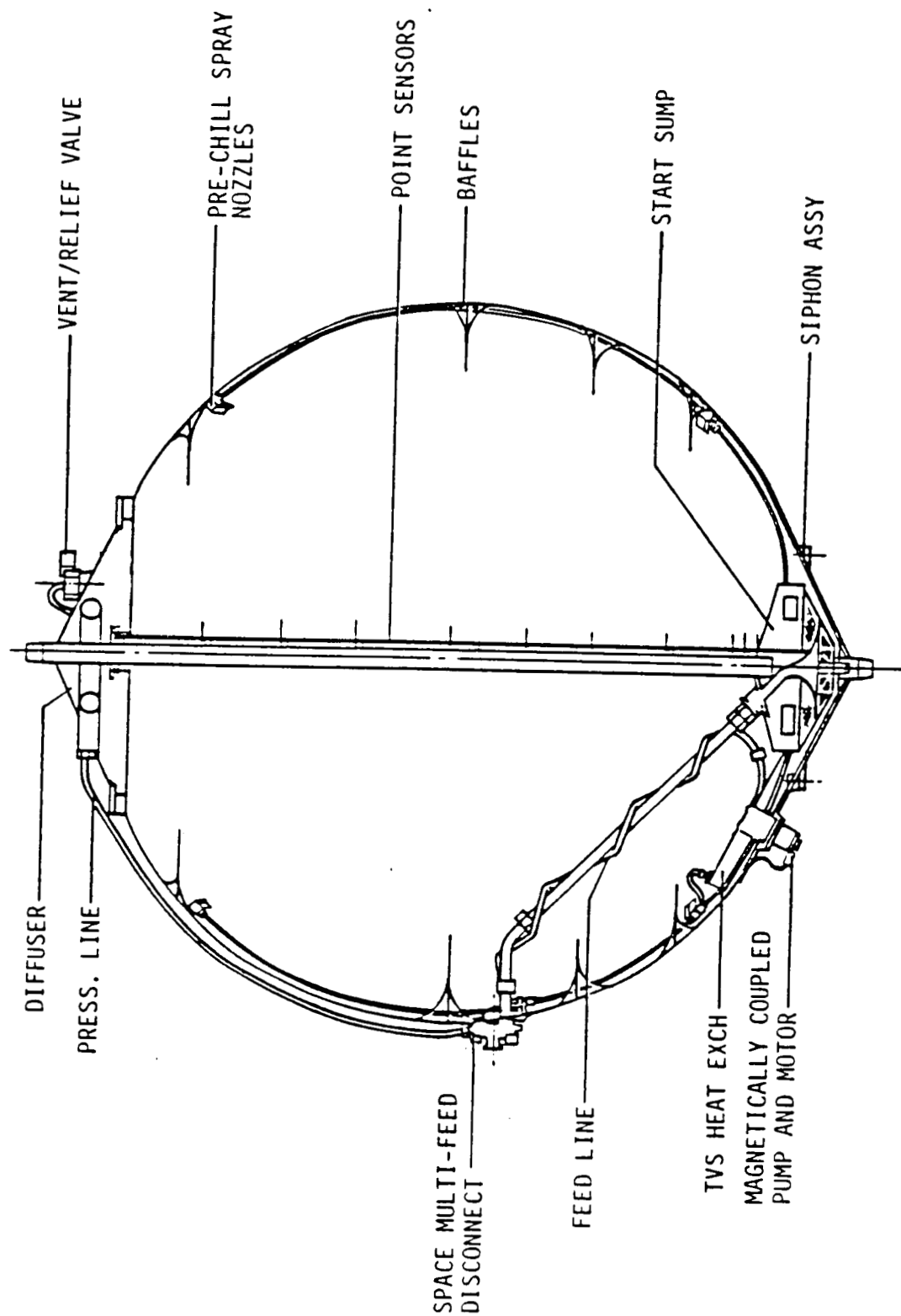
References -

1. Blevins D., J. Henderson, C.Hohmann, W. Karakulka, and Ralph J. Taeuber, <u>Design Evolution of the Orbiter Reaction Control Subsystem</u>. NASA Technical Report N85-16944, 1985.

2. Epps R.C., and F.E. Hughes, <u>Reaction Control System Training Manual of the Space Shuttle</u>. NASA - MSFC, April 1987.

3. <u>Manned GEO Sortie Study</u>. NASA - JSC, 1983.

4. North D.D., <u>State-of-the-Art Report : Chemical Rocket Propulsion</u>. Dept of Aero and Mech Engr, Univ of Virginia, November 1986.

5. <u>Orbital Transfer Vehicle - Configuration and Subsystem Trade Studies</u> (1986). The Boeing Aerospace Company.

6. <u>Orbital Transfer Vehicle - Propulsion Subsystems</u>, Vol 5, (1985). The Martin Marietta Aerospace Company.

7. <u>Orbital Transfer Vehicle - Concept Definition and System Analysis Study, RCS Selection.</u> 30 November 1984. The Martin Marietta Company.

APPENDIX

COMPUTER PROGRAM USED TO DETERMINE
TANK SIZES FOR REQUIRED MPS PROPELLANT

```
!                              by Miles O. Duquette
!
!
!                      for AE 442, Space Vehicle Design
!
!                              10 February 1988
DIM R(0:150)
SET MODE "EGAHIRES"     ! This program requires an EGA monitor.
OPTION NOLET

                       ! The next 8 lines initialize the variables:

ISP=880                ! ISP = specific impulse
RHO=4.42               ! RHO = density of liquid hydrogen
DV=28000               ! DV = the total delta V's
MS=20000               ! MS = mass of the spacecraft (structure only)
MPL=15000              ! MPL = mass of the payload
ME=5000                ! ME = mass of the engine
KT=0.05447             ! KT = ratio of mass of tanks to mass propellant
EXP=2.718281828        ! EXP = the number 'e'

  UNTIL choice=9       ! start main program loop
  choice=0
  SET COLOR 12         ! set color to bright red
  FOR I=0 TO 150
    R(I)=0             ! intialize array R with zero's
  NEXT I
  CLEAR
  SET CURSOR 6,22
  PRINT "PLEASE SELECT ONE OF THE FOLLOWING:"
  SET COLOR 14                                 ! set color to yellow
  SET CURSOR 9,17
  PRINT "<1> ";
  SET COLOR 3                                   ! set color to cyan
  PRINT " Perform computation w/ present data."
  SET COLOR 14
  SET CURSOR 11,17
  PRINT "<2> ";
  SET COLOR 3
  PRINT " View or Change the present data."
  SET COLOR 14
  SET CURSOR 13,17
  PRINT "<3> ";
  SET COLOR 3
  PRINT " Compute required tankage from present data."
  SET COLOR 14
  SET CURSOR 17,17
  PRINT "<9> ";
  SET COLOR 13                 ! set color to bright magenta
  PRINT " Exit this program."
  SET COLOR 10                 ! set color to bright green

  DO UNTIL (choice>0 AND choice<4) OR choice=9
    SET CURSOR 22,27
    INPUT PROMPT "Your selection: ", choice
```

```
IF choice=1 THEN          !  choice 1 is to do computations

  KM = 1 - 1/EXP^(DV/(ISP*32.174))     ! compute the ratio Mp/Mo
  MO = (MS+MPL+ME)/(1-(KM+KT*KM))       ! compute the mass Mo
  MP = KM*MO                            ! compute the mass of propellant, Mp
  VP = KM*MO/RHO                        ! compute the volume of propellant

  CLEAR                   !  now show the results
  SET COLOR 13
  BOX AREA 0.36,0.6,0.6,0.8
  SET COLOR 14
  SET CURSOR 8,35
  PRINT " RESULTS "
  SET COLOR 3
  SET CURSOR 14,20
  PRINT "TOTAL MASS, MO  =  ";
  SET COLOR 15                              ! set color to bright white
  PRINT USING "###,###":MO;
  SET COLOR 3
  PRINT " lb"
  SET CURSOR 16,20
  PRINT "MASS OF PROPELLANT, MP  =  ";
  SET COLOR 15
  PRINT USING "###,###":MP;
  SET COLOR 3
  PRINT " lb"
  SET CURSOR 18,20
  PRINT "PERCENT PROPELLANT, MP/MO  =  ";
  SET COLOR 15
  PRINT USING "###.##":KM*100;
  SET COLOR 3
  PRINT " %"
  SET CURSOR 20,20
  PRINT "VOLUME OF PROPELLANT, VP  =  ";
  SET COLOR 15
  PRINT USING "##,###":VP;
  SET COLOR 3
  PRINT " ft^3"
  SET CURSOR 24,1
  PRINT "Press any key to continue...";
  GET KEY wait

ELSEIF choice=2 THEN  !  choice 2, view/change paramters
  done=0
  DO
  RESTORE              !  reset 'read' pointer to beginning of data

  CLEAR                !  show present settings
  SET COLOR 12
  SET CURSOR 4,22
  PRINT "THE OTV PARAMETERS ARE AS FOLLOWS:"
  FOR X=1 TO 4
```

```basic
  SET COLOR 14
  READ num1$,text1$,use1$,var1,unit1$,num2$,text2$,use2$,var2,unit2$
  PRINT num1$;
  SET COLOR 3
  PRINT text1$;
  SET COLOR 15
  SELECT CASE var1
  CASE 1
    value = ISP
  CASE 2
    value = RHO
  CASE 3
    value = DV
  CASE 4
    value = KT
  END SELECT
  PRINT USING use1$:value;
  SET COLOR 3
  PRINT unit1$;
  SET COLOR 14
  PRINT num2$;
  SET COLOR 3
  PRINT text2$;
  SET COLOR 15
  SELECT CASE var2
  CASE 5
    value = MS
  CASE 6
    value = MPL
  CASE 7
    value = ME
  CASE 8
    EXIT FOR
  END SELECT
  PRINT USING use2$:value;
  SET COLOR 3
  PRINT unit2$
NEXT X
DATA "<1> ","Specific Impulse: "," ###",1," sec          ","      <5> "
DATA "Mass of spacecraft: "," ##,###",5," lb"
DATA "<2> ","Density of Hydrogen: "
DATA "#.##",2," lb/ft^3","      <6> ","Mass of payload: "," ##,###",6
DATA " lb","<3> ","Total Delta V's: "," ##,###",3," ft/s      ","    <7> "
DATA "Mass of engine: "," ##,###",7," lb","<4> ","Tank to prop. ratio: "
DATA " %.####",4," lb/lb","","","",8,""
SET COLOR 3
SET CURSOR 18,15
PRINT "Is there anything you'd like to change? ";
GET KEY chg
chg$=UCASE$(chr$(chg))
PRINT chg$
IF chg$="Y" THEN
  sel=-1
  DO UNTIL
```

```
          SET CURSOR 20,15
          INPUT PROMPT "Which number do you want to change? (0 to abort) ": sel
       LOOP
       IF sel<>0 THEN                    .
          SET CURSOR 23,15
          INPUT PROMPT "What will its new value be? ": newval
          SELECT CASE sel
            CASE 1
              ISP = newval
            CASE 2
              RHO = newval                    ORIGINAL PAGE IS
            CASE 3                             OF POOR QUALITY
              DV = newval
            CASE 4
              KT = newval
            CASE 5
              MS = newval
            CASE 6
              MPL = newval
            CASE 7
              ME = newval
          END SELECT
       END IF
       done=0
     ELSE
       done=1
     END IF
   LOOP UNTIL done=1


ELSE                        ! choice 3, compute tankage requirements

   IF choice=9 THEN EXIT DO      ! check for 'end program' choice

   IF VP=0 THEN                     ! make sure computations have been done
     SET CURSOR 12,15
     SET COLOR 20
     PRINT "YOU HAVE NOT RUN THE CALCULATIONS YET"
     SET COLOR 3
     SET CURSOR 22,10
     PRINT "      Press any key to continue....";
     GET KEY wait
   END IF
   IF VP=0 THEN EXIT IF

   SET COLOR 5                      ! set color to magenta
   DS,DC,H=0
   DO WHILE DS=0
     SET CURSOR 10,55
     PRINT "      "
     SET CURSOR 10,10
     INPUT PROMPT "What diameter, in feet, for spherical tanks? ":DS
   LOOP
   DO WHILE DC=0 OR H=0
```

```
      SET CURSOR 13,10
      INPUT PROMPT "Enter the diamter and height for cylindrical tanks: ":DC,H
LOOP
SET CURSOR 23,20
PRINT "One moment please..."
                                        ! do computations of tankage
VS=4*PI*(DS/2)^3/3
VC=PI*(DC/2)^2*H + 4*PI*(DC/2)^3/3
NS=INT(VP/VS+0.99)
NC=INT(VP/VC+0.99)
IF VS>VC THEN
   T1=VS
   T2=VC
   N1=NS
ELSE
   T1=VC
   T2=VS
   N1=NC
END IF
FOR I=N1 TO 0 STEP -1
   N2=-1
   DO
     N2=N2+1
     VT=I*T1+N2*T2
   LOOP UNTIL VT>=VP
   R(I)=VT-VP
NEXT I
BEST=999999
FOR I=0 TO N1
   BEST=min(BEST,R(I))
   IF BEST=R(I) THEN BC=I
NEXT I
IF VS>VC THEN
   N1=BC
   N2=INT((VP-N1*T1)/T2+0.99)
   VAVG=(N1*T1+N2*T2)/(N1+N2)
ELSE
   N2=BC
   N1=INT((VP-N2*T1)/T2+0.99)
   VAVG=(N1*T2+N2*T1)/(N1+N2)
END IF


CLEAR              !  show results of calculations
SET COLOR 12
SET CURSOR 1,20
PRINT "POSSIBLE TANKAGE CONFIGURATIONS"
SET COLOR 3
SET CURSOR 3,17
PRINT "(Volume of propellant = ";
SET COLOR 6                          ! set color to brown
PRINT USING "##,###.#":VP;
SET COLOR 3
PRINT " ft.^3)"
```

```
PRINT "Sphere Diameter:  ";
SET COLOR 6
PRINT USING "###.#":DS;
SET COLOR 3
PRINT " ft"
SET CURSOR 7,16
PRINT "Cylinder Diameter, Height: ";
SET COLOR 6
PRINT USING "###.#, ##.#":DC,H;
SET COLOR 3
PRINT " ft"
SET CURSOR 10,1
PRINT "                                    VOLUME PER      # UNITS"
PRINT "                      SHAPE         UNIT (ft^3)     REQUIRED"
SET COLOR 6
FOR I=1 TO 4
  SET CURSOR 12+(I-1)*3,1
  PRINT "            -------------------------------------------------"
NEXT I
FOR I=1 TO 12
  SET CURSOR 9+I,31
  PRINT "¦"
  SET CURSOR 9+I,47
  PRINT "¦"
NEXT I
SET COLOR 4                  ! set color to red
SET CURSOR 14,16
PRINT "SPHERICAL"
SET CURSOR 17,15
PRINT "CYLINDRICAL"
SET CURSOR 19,15
PRINT "combination"
SET CURSOR 20,11
PRINT "SPHERES & CYLINDERS"
SET CURSOR 19,34
PRINT "Average Vol:"
SET CURSOR 19,52
PRINT "Sph:"
SET CURSOR 20,52
PRINT "Cyl:"
SET COLOR 15
SET CURSOR 14,36
PRINT USING "##,###.##":VS
SET CURSOR 17,36
PRINT USING "##,###.##":VC
SET CURSOR 14,53
PRINT NS
SET CURSOR 17,53
PRINT NC
SET CURSOR 20,36
PRINT USING "##,###.##":VAVG
SET CURSOR 19,57
PRINT N1
```

```
    SET CURSOR 22,50
    PRINT "Remainder: ";
    PRINT USING "#,###.#":R(BC)
    SET CURSOR 24,2
    PRINT "Press any key to continue..."
    GET KEY waitagain
  END IF
LOOP
SET COLOR 15
CLEAR
SET CURSOR 12,30
PRINT "Press any key..."
END
```

IX. SERVICING SYSTEM

Introduction

The OTV will have two basic capabilities in Earth orbit:
1) transporting payloads between Low Earth Orbit (LEO) and
Geosynchronous (GEO) Orbit and 2) servicing spacecraft in GEO
orbit. The operations of 2) are designated in-situation (in-
situ) because the satellite is serviced in its orbit. This
report presents the design of the servicing system necessary to
fulfill these OTV functions.

The primary mission of the OTV Servicing System is to extend
the lifetimes of GEO spacecraft that might be cut short by
"infant mortality", random failures, and expendables (or
consumables) exhaustion. In fulfilling this mission, the
Servicing System must execute the following on-orbit servicing
functions:

-- resupply expendables, primarily Attitude Control System
    propellants and water,

-- restore orbital spacecraft systems to full operational
    capability by replacing failed elements, and

-- upgrade spacecraft systems to incorporate advances in
    technology.

The U.S. Space Station, located in LEO, will serve as a base of
operations for the OTV. Should a satellite or platform need
servicing equipment, parts, or man-hours that the OTV is
incapable of providing, the OTV will have the capacity to
retrieve the spacecraft and transport it back to the space
station for repair. After repairs are completed, the OTV has the

155

capability to return the satellite or platform to its geosynchronous position.

System Requirements

The OTV Servicing System is fundamentally constrained to accommodate the requirements of the GEO customer in the 1990s. Specifically, the design of the Servicing System is based on these factors:

-- spacecraft mass (dry),

-- dimensions,

-- consumables requirements, and

-- hardware design.

The analysis of the these factors concentrates on the design of commercial satellites. Other users of GEO orbits are NASA and the Department of Defence (DoD). Currently about 70 commercial satellites are located in GEO orbit, providing television, telegraph, and data transmissions [5, p. 42]. Approximately 150 commercial satellites will be in GEO orbit by 1990. This predominance of the commercial satellite warrants its emphasis in the design of the Servicing System.

Three different commercial satellite designs are in existence today and will be utilized into the next decade. The three types are:

-- spin stabilized satellites,

-- three-axis stabilized satellites, and

-- three-axis, hybrid satellites.

The trend in satellite design is towards large platforms with

increased numbers of communications transponders. A platform is classified as any satellite with a total mass exceeding 5,000 lb. Table 1 summarizes the design characteristics of the three satellite types and a proposed platform, the Geostationary Platform. The Geostationary Platform is intended to support communications and maritime payloads.

	Spinner	3-Axis	Hybrid	Platform
Baseline Satellite	Hughes HS-393	RCA K2	Ford FS-1300	Geostationary Platform
Design Life (years)	10	10	10	7
Replacement Cost ($M, 1985)	54.2	50.9	64.6	175
Stowed Size (feet)	Dia = 11.94 Len = 10.99	5.15 x 7.15 x 5.81	8.2 x 6.17 x 8.66	Dia = 110 Len = 164 *
Mass (pounds)	3,611	2,877	4,149	12,000
Propellant Type	Bi-prop.	Hydrazine	Bi-prop.	Hydrazine

*on-orbit, unstowed

Table 1: Satellite/Platform Characteristics

A GEO satellite is typically designed for an operational life of 7 to 10 years. At the end of this period, the satellite has exhausted its Attitude Control System (ACS) propellant supply. ACS propellant is an example of consumables required by the satellite or platform in order to function. The

157

Geostationary Platform will burn 1,764 lb of propellant over its 7 year lifetime.

The Geostationary Platform ACS makes use of hydrazine propellant, commonly used in spacecraft designs. Bi-propellants have been recently incorporated into satellite ACS designs. Bi-propellants offer higher specific impulses. The Space Station and LEO OMV designs make use of bi-propellants, serving as a catalyst for standardization of stored propellants. Yet, the fact that the proposed platform will utilize hydrazine propellant, indicates that the OTV must be capable of resupplying both propellant types.

Another factor contributing to the effective lifetime of a satellite is the occurrence of component malfunctions. The NASA Solar Maximum Mission demonstrates how the lifetime of a satellite can be cut short by component failure. The Solar Max satellite was launched on February 14, 1980. Soon after launch three of four fuses blew in the ACS, crippling the satellite's ability to maintain the correct orbital attitude. This satellite, however, was designed with servicing in mind. The Solar Max satellite is an example of the Multi-Mission Spacecraft (MMS) design concept. Figure 1 shows the Solar Maximum observatory.

The MMS is a standardized reusable space platform capable of supporting a wide variety of Earth-obit programs. The replacement of MMS hardware components is facilitated by modularity in its design. The basic structure of the MMS supports modularized power, communications, data handling, and

altitude control components, which may be replaced in order to
prolong the usefulness of the spacecraft. The individual modules
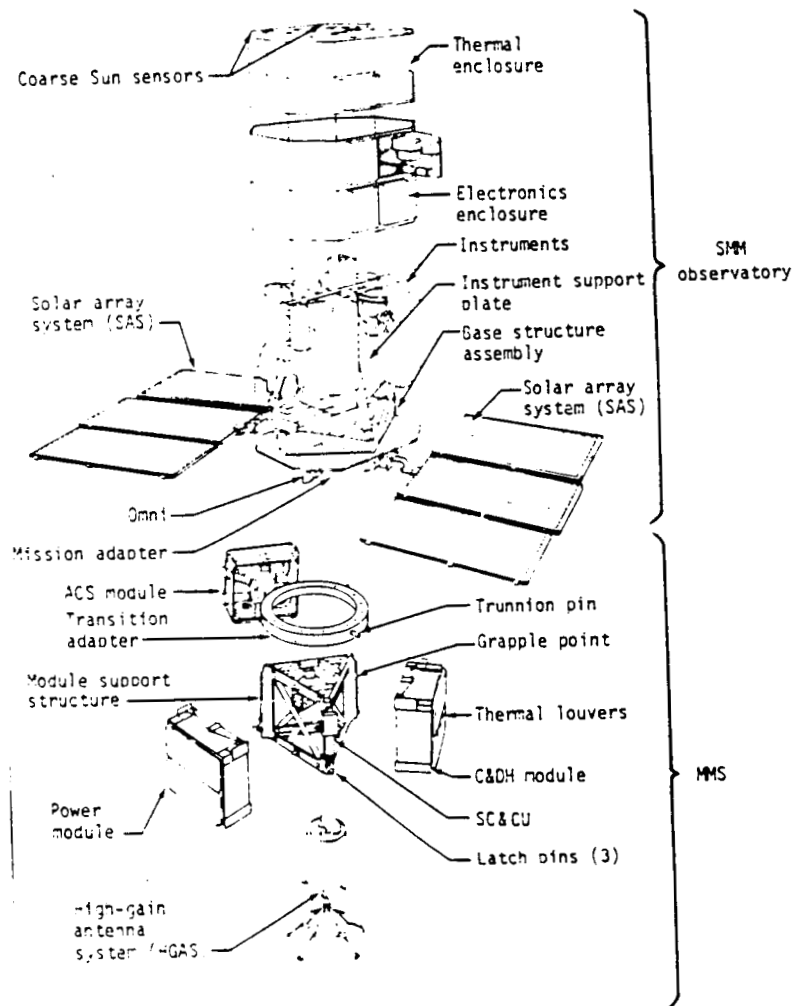are commonly designated Orbital Replacement Units or ORUs.

Figure 1: The Solar Maximum Observatory

The ORUs are held in place by two jack-screws which require
maximum torques of 70 to 90 ft-lbs. An electrically powered tool

159

called the Module Servicing Tool (MST) latches to the module and provides the necessary torque to mate and demate the module from the module support structure. The MST is shown in Figure 2. The module retention system also mates/demates the electrical connections between the MMS and the module.
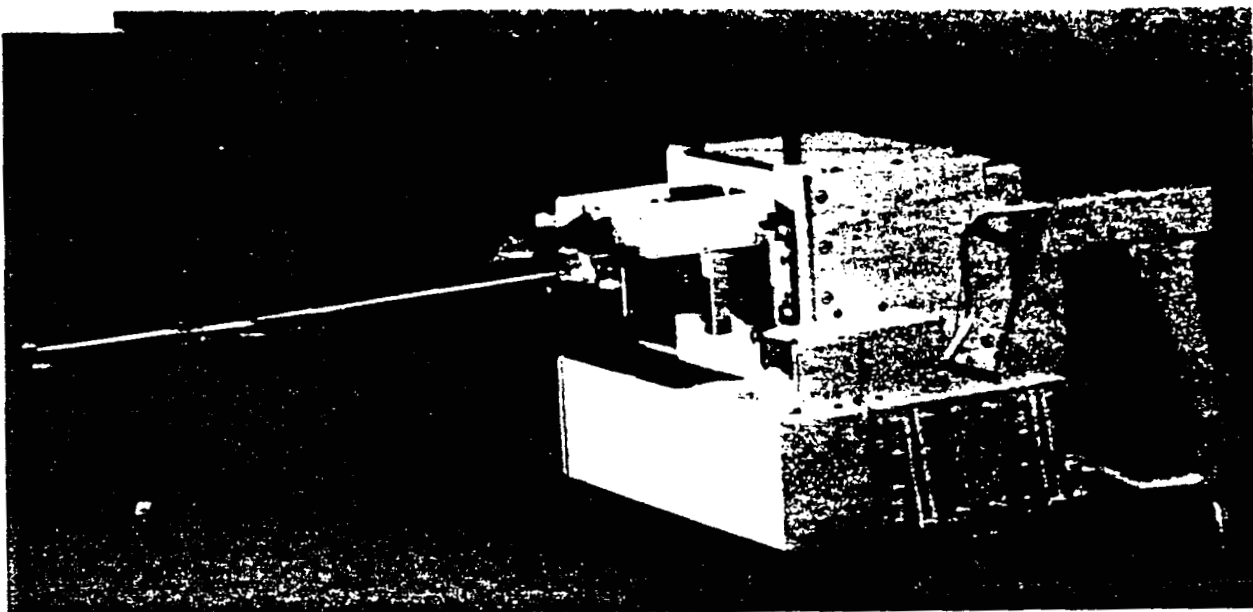

Figure 2: Module Service Tool

Many existing designs do not utilize modularization and serviceable ORUs. Current satellites lack standardized mechanical interfaces for grappling (affixing a manipulator end effector to the structure of the satellite), berthing, and fluid resupply. Such spacecraft would require highly complex servicing operations. In fact, it may not be feasible to (in-situ) service non-standard satellites. These satellites could be transported to the Space Station for servicing if cost effective.

160

Servicing in GEO orbit will be practical only if satellites are designed with modular hardware and standardized fixtures for manipulation. The large platforms that will be placed in GEO orbit in the 1990s will represent larger investments that must be protected and sustained through in-situ servicing. The OTV Servicing System is designed for the servicing of MMS type spacecraft.

Servicing Operations

After the OTV rendezvous with the GEO satellite, these operations take place: Capture and Berthing, Consumables Resupply, ORU Exchange, and Non-Standard Repairs. The following section briefly describes each operation.

-- _Capture and Berthing_ the satellite for servicing on the OTV. Capture can be complicated by the attitude and outfitting of the satellite. A cooperative satellite is favorably positioned for capture by the Remote Manipulating System (RMS) or robot arm; it does not require any special stabilization or de-tumbling. In some cases the satellite can employ its own propulsion system to position itself for capture. The satellite is already fixed with an RMS grapple fixture, allowing the RMS to grip the satellite without causing damage to its structure.

An uncooperative satellite must first be stabilized before final rendezvous and capture. A satellite lacking a standard grappling fixture must be captured using specially designed equipment. An Extravehicular Activity (EVA) excursion unit, the Manned Maneuvering Unit (MMU), was used in the Solar Max repair

mission of STS-41C to stabilize the satellite's angular spin rate about one axis. The MMU is a self-contained, propulsive backpack that provides mobility to the unpropelled crewmember during EVA (those operations outside the pressurized environment of the vehicle).

The Shuttle RMS can not grapple the satellite until it is essentially stationary with respect to the vehicle. Once stabilized, the RMS can grapple the satellite as long as it is equipped with a compatible grappling fixture. Multi-axis tumbling will require, as yet, undeveloped technology.

Berthing of the satellite to the Flight Support Station (FSS) is teleoperated by means of the RMS. MMU stabilization/RMS grappling is the baseline mode of satellite capture and berthing in the OTV Servicing System.

-- Expendables Resupply: the resupply of ACS propellants, pressurants, liquid helium, and water to the orbital spacecraft. Fluid propellants, pressurants, and liquid helium are transferred via umbilical connection. Once the umbilical connection has been established, the resupply is controlled automatically from within the OTV. Water will be resupplied by ORU exchange.

-- ORU Exchange: the replacement of a failed or obsolete ORU on a MMS with a functioning or improved unit via manual EVA. The ORU to be replaced is removed by means of the MST and temporarily stowed at a parking position. The new ORU is then unstowed and installed in the satellite/platform. This operation is performed by one or two EVA crewmembers supported by the intravehicular RMS operator. The Manipulator Foot Restraint (MFR), shown in Figure

3, is a portable workstation, allowing an EVA crewmember to
access worksites within reach of the RMS. The MFR provides
restraint for one crewmember, tool storage, and the transport of
large ORUs.

ORIGINAL PAGE IS
OF POOR QUALITY
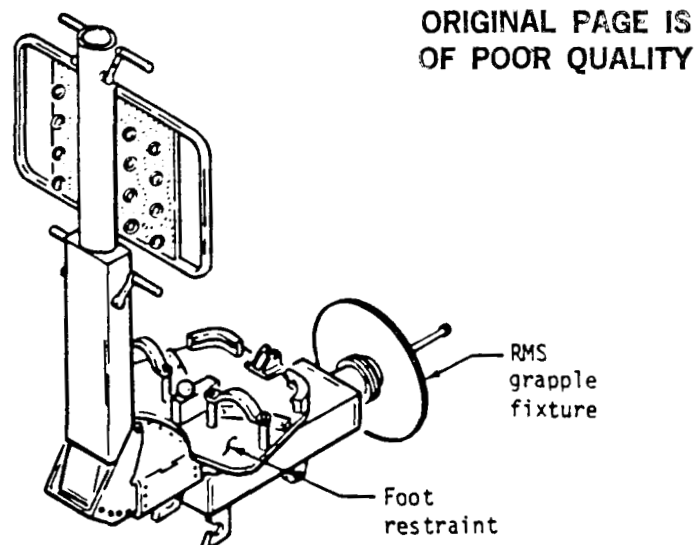
RMS
grapple
fixture

Foot
restraint

Figure 3: Manipulator Foot Restraint

-- Non-Standard Repair: maintenance operations that do not
involve ORU exchanges. These operations involve those tasks
required to restore a hardware component in a degraded state of
performance to a state acceptable to the operational requirements
of the system. The range of repair operations is unlimited, yet
two frequently occurring repair operations are: surface cleaning
and freeing jammed mechanisms. Sensors and other sensitive
instruments are extremely prone to contamination from thrus' r
plume debris, requiring period cleaning.

163

Repair operations necessitate in-situ interpretive skills and non-standard work sites. These operations require a high level of crew interaction with the intravehicular crew and ground control. Specialized tools, such as those for removing jams in Figure 4, are required in repair operations.
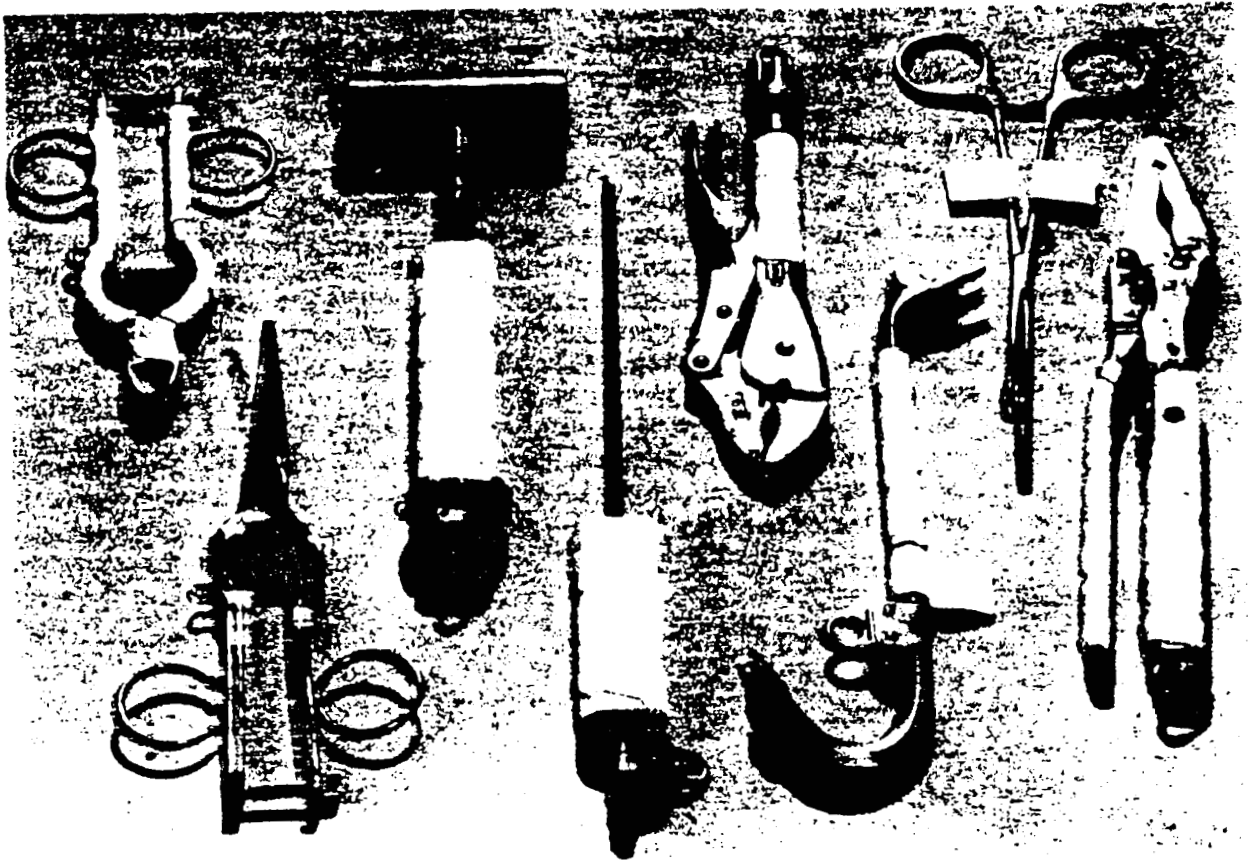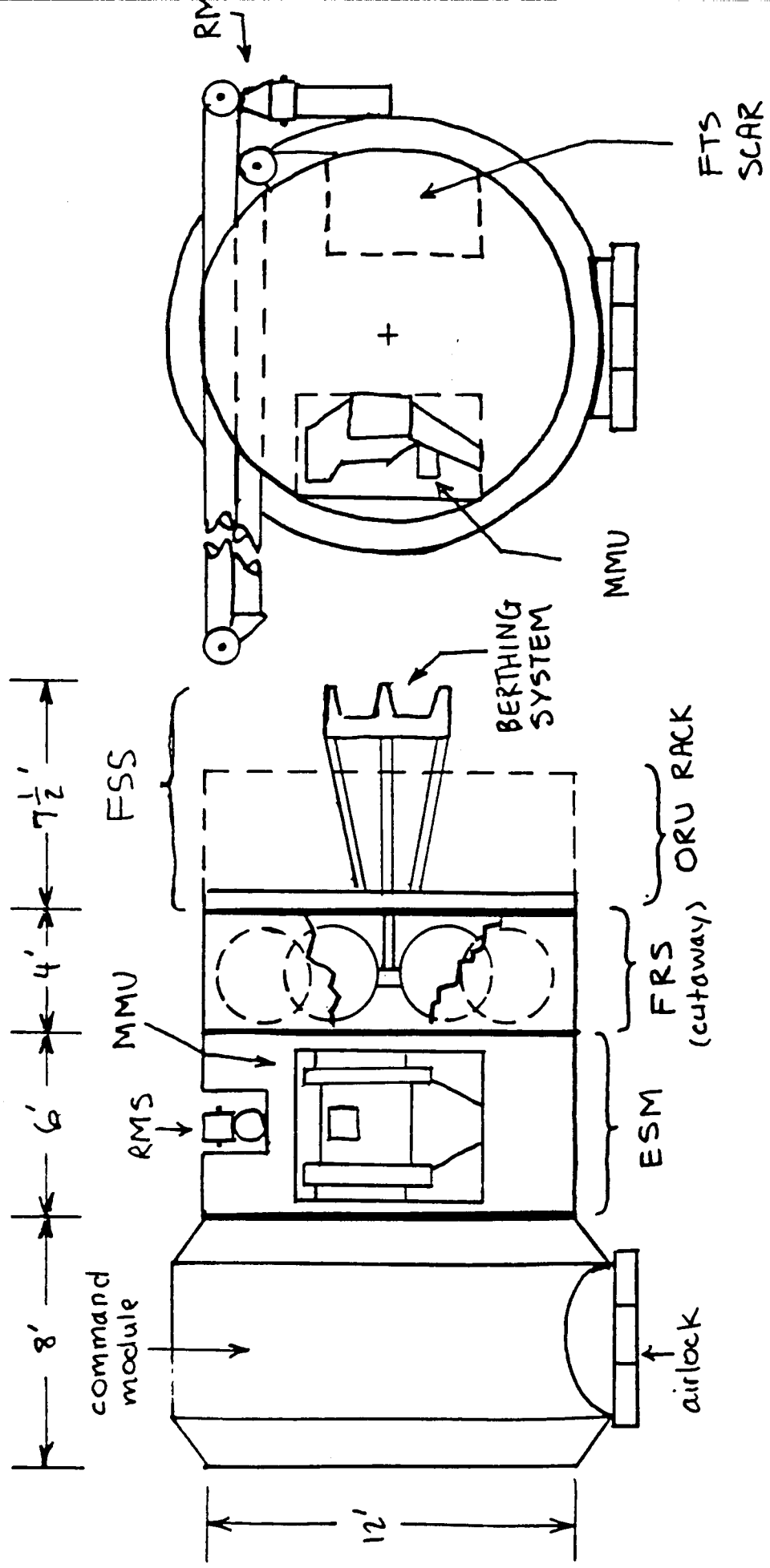

Figure 4: EVA Jam Removal Tools

Elements of the Servicing System

The following section presents the overall design of the
Servicing System and its major elements, which have already been
discussed in the Servicing Operations section of this chapter.
Figure 5 shows the design of the Servicing System. The system
consists of three modular elements: the EVA Support Module,
Fluid Resupply System, and the Flight Support Station. These
elements are independent of each other secured by latch
mechanisms. Modularity of the Servicing System allows simplified
servicing of its elements and the potential for upgrading
components.

1.   The EVA Support Module (ESM)

The ESM primarily provides the structural support for the
OTV RMS and the MMU and its flight station. The OTV RMS is the
same RMS as used on the STS Orbiter. The RMS is shown in Figure
6. The total length of the RMS is approximately 50 ft. This
length allows the OTV to safely grapple satellites at a distance
of over 30 ft. away. This capability is valuable in conserving
reaction control propellants that would be spent in positioning
the OTV within close proximity of the satellite position. This
length, however, presents a special problem in stowing the robot
arm during vehicle accelerations. The solution is to stow the
arm within a notch in the side of the ESM, shown in Figure 5.

OTV SERVICING SYSTEM

Labels: RMS, FTS SCAR, MMU, BERTHING SYSTEM, FSS, FRS (cutaway), ORU RACK, ESM, MMU, RMS, command module, airlock

Dimensions: 7½', 4', 6', 8', 12'

WRIST YAW

WRIST ROLL

SHOULDER YAW

WRIST PITCH

ELBOW PITCH

SHOULDER PITCH

JOINT REFERENCE
SYSTEM

277 95 IN
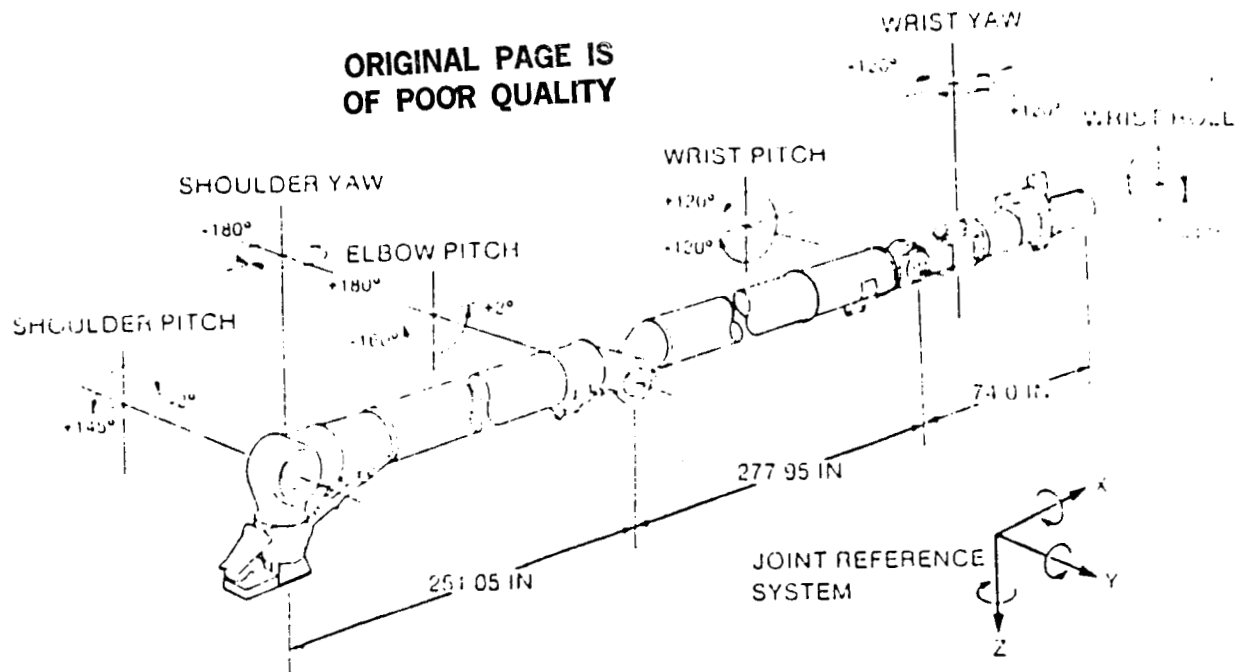
Figure 6: Remote Manipulator System (STS)

GN₂ Pressure Gage (2)

Thruster Cue (2)

Thrusters (24)

MMU PLSS Release Ring (2)

PLSS Contact Button (4)

Main Power Switch (2)

Battery (2)

Locator Light Switch

GN₂ Cross Feed/ Recharge Valve (2)

PLSS Latch (2)

GN₂ Tanks (2)

Arm Stowage Lock

Circuit Breakers (3)

Ancil Equip. Attach Points (3 Each Side)

Rotational Hand Controller

Propellant Recharge QD (2)

Recharge Tether Bracket (2)

Adjustable Arm (2)

GSE Attach Points (4 Each Side)

Arm Hinge and Lock

EVA Power Switches A and B

External Power Connector

External Power Switches

Locator Light (3)

Translational Hand Controller

Gyro Power Switch

Arm Angle Adjustment Lever (2)

Alternate Control Mode Switch

Arm Length Adjustment Lever (2)
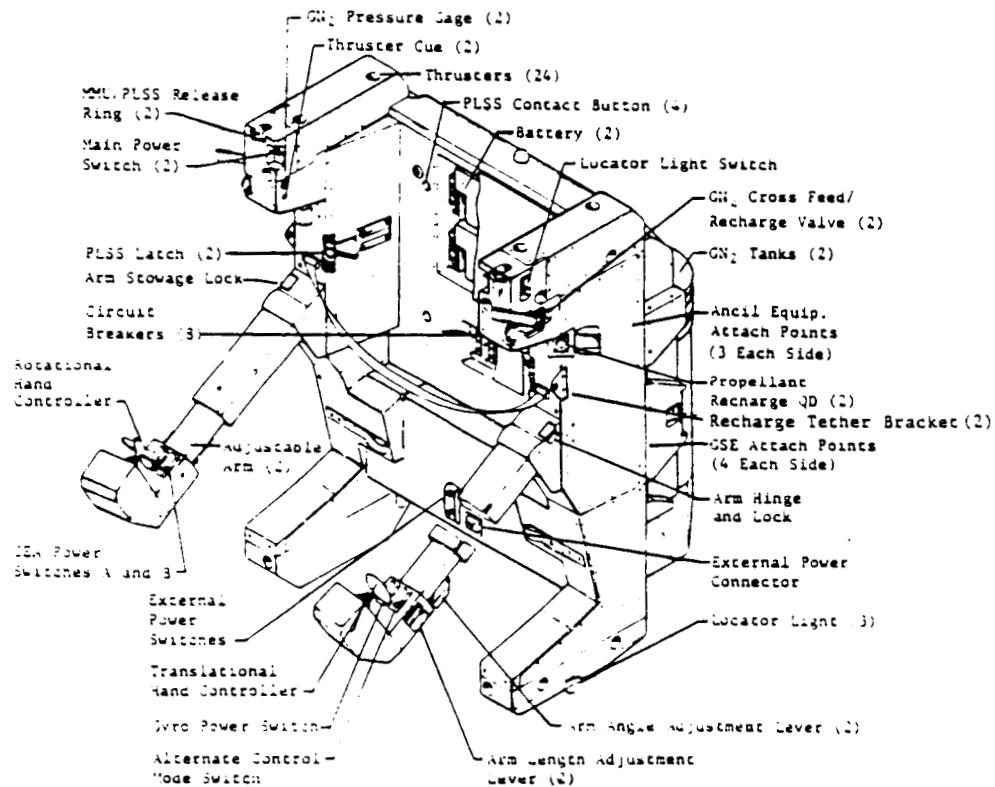
Figure 7: The Manned Maneuvering Unit

167

The MMU is a self-contained, propulsive backpack designed to provide mobility to the unpropelled astronaut within an EVA Mobility Unit or EMU. Figure 8 shows an EVA crewmember suited up in an EMU and having donned a MMU. The MMU provides six-degree-of-freedom control authority along with piloting logic in order to execute a full range of translations, rotations, or combinations thereof. The MMU flight station provides structural support for the unit and stores gaseous nitrogen propellant (GN_2) for reservicing.

Figure 8: Astronaut in EMU and MMU

Note that The longitudinal path from the Command Module to the FSS has been left free to facilitate the translation of EVA crewmembers along the OTV exterior.

2. The Fluid Resupply System (FRS)

The FRS executes expendables resupply operations. The FRS module has room for up to six, 45 in. diameter spherical tanks developed for the STS Orbiter Reaction Control System. These six tanks are currently in production. This volume will accommodate up to 8,000 lb. of propellant, satisfying any one platform's needs well into the twenty-first century [5, p.33]. Depending on the particular mission, the propellant can be hydrazine or bi-propellant.

Nominally, the berthing of the satellite will automatically establish fluid transfer connections. However, to accommodate variations in satellite design, an umbilical system must be available. The umbilical connection will be connected via manual EVA. Plumbing connections are concurrent with the mechanical interface between the FRS and the FSS.

3. The Flight Support Station

The FSS serves as a satellite workstation allowing the satellite to be secured and manipulated for servicing. The location of the FSS at the front of the OTV does not limit the size of the platform or satellite to be serviced. The positioning of the FSS along the longitudinal axis of the OTV avoids off-center mass distributions.

The FSS provides EVA mobility and positioning aids, such as interfaces for portable foot restraints. Automatic interfaces

of the FSS provide propellant resupply and power to the berthed satellite as well as component diagnosis, testing, and checkout. The base of the FSS is a rack for ORU storage, providing structural support and power to each ORU. Thermal control is not necessary since each ORU can be expected to have an independent system.

Technological Growth

The OTV Servicing System has been "scarred" for technological growth. Manual EVA is currently the only means of performing operations such as ORU exchanges and satellite capture. For this reason manual EVA is the baseline mode of execution for servicing operations requiring a high degree of dexterity. However, a robotic system is under study to automate and augment EVA tasks. This system, the Flight Telerobotic Servicer (FTS), will be a multi-purpose robot, teleoperated from within the OTV Command Module as the RMS will be.

The Goddard Space Flight Center has produced a preliminary design, designated the Strawman. The Strawman telerobot has two dexterous manipulators and one manipulator to be used for attaching the unit to a worksite. The Strawman (FTS) concept is shown in Figure 9. The Strawman has grappling fixtures so that the RMS can position the unit in the same way an astronaut is positioned for EVA via the MFR. Cameras and light systems mounted on extended booms relay images of the worksite back to the intravehicular operator. Figure 10 shows how the MST can be adapted for robotic use.

The ESM has sufficient volume to accommodate the FTS. The FTS should be positioned under the RMS shoulder joint to facilitate grappling. The FTS will require its own flight station as does the MMU.

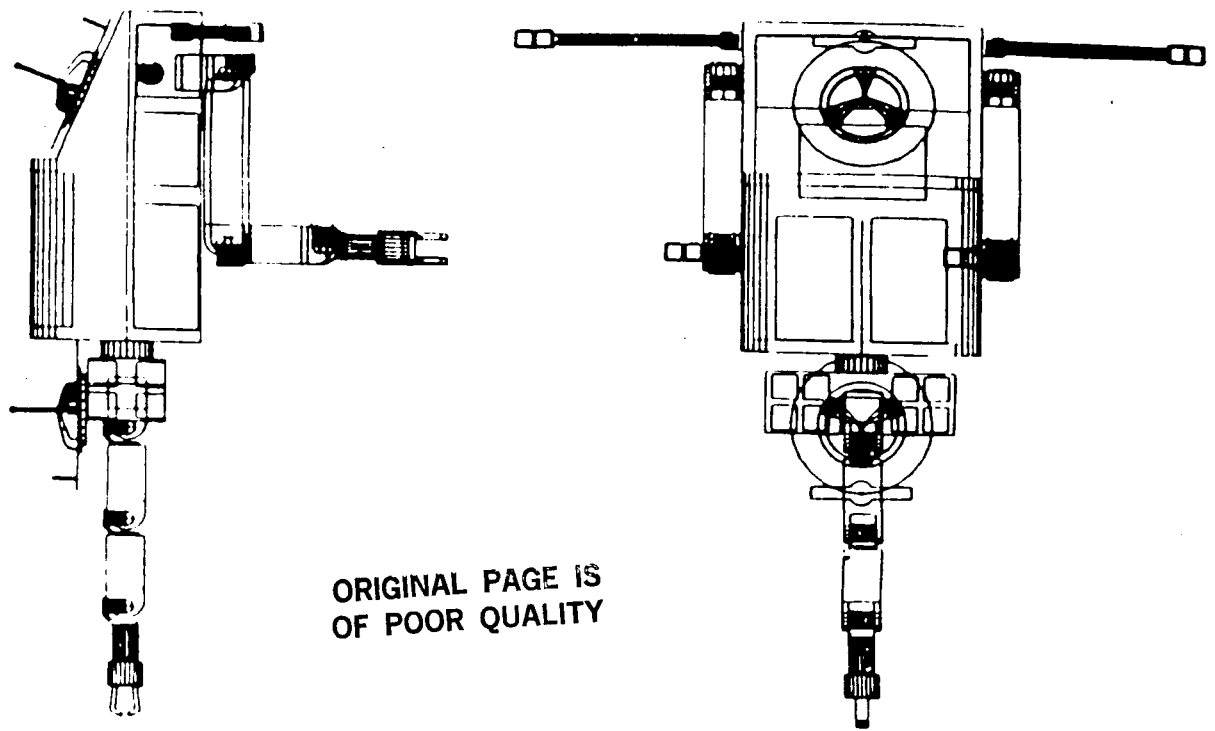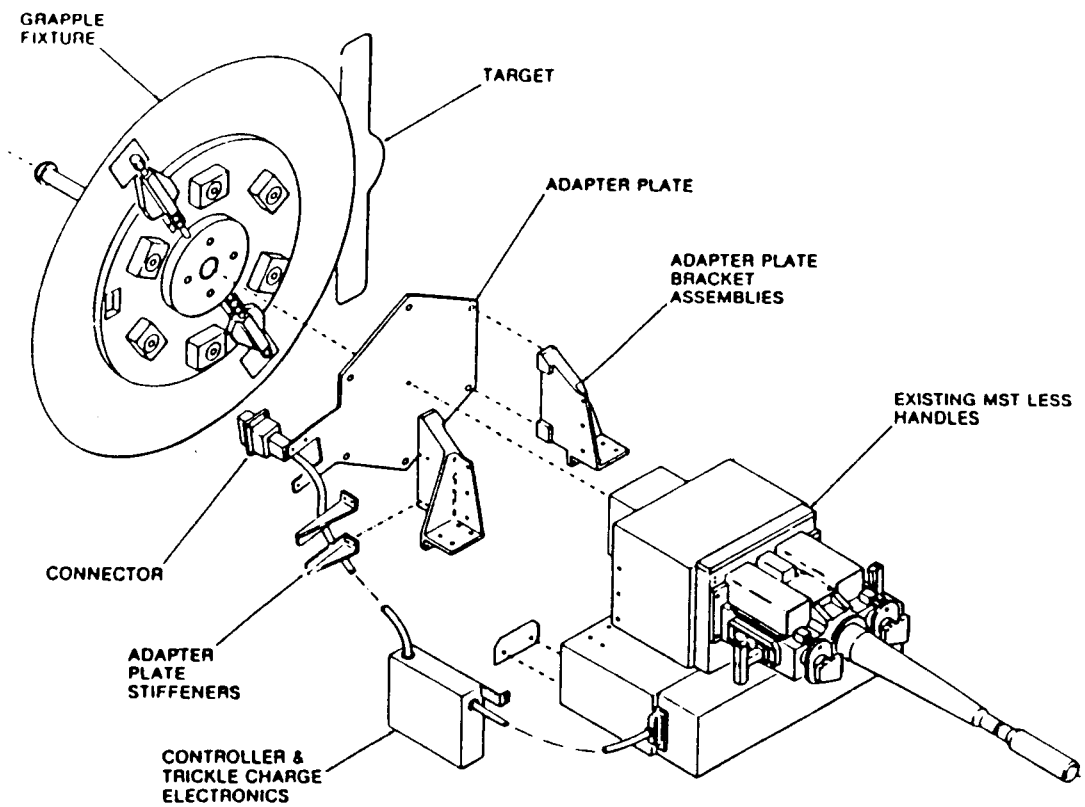Figure 9:   Strawman Concept

GRAPPLE
FIXTURE

TARGET

ADAPTER PLATE

ADAPTER PLATE
BRACKET
ASSEMBLIES

EXISTING MST LESS
HANDLES

CONNECTOR

ADAPTER
PLATE
STIFFENERS

CONTROLLER &
TRICKLE CHARGE
ELECTRONICS

Figure 10:   RMS Module Service Tool

172

REFERENCES

[1]   Martin Marietta Aerospace, <u>Space Station Definition and
      Preliminary Design</u>, NASA, Johnson Space Center,
      Houston, Texas, Feb. 1987.

[2]   Rouen, Michael, and Woods, Thomas, <u>Advanced EVA System
      Design Requirements Study</u>, NASA, Johnson Space Center,
      Houston, Texas, Jan. 1986.

[3]   <u>Satellite Servicing Mission Preliminary Cost Estimation
      Model</u>, NASA, Johnson Space Center, Houston, Texas,
      1986.

[4]   <u>Satellite Servicing Pricing Estimation Instruction Booklet</u>,
      NASA, Johnson Space Center, Houston, Texas, 1986.

[5]   <u>Space Platform Expendables Resupply Concept Definition
      Study</u>, Final Report, Rockwell International, Jan. -
      Oct., 1985.

[6]   <u>Space Transportation System: EVA Description and Design
      Criteria</u>, NASA, Johnson Space Center, Houston, Texas,
      May 1983.

X.   SHIELDING/STRUCTURES

Prolonged space travel presents many risks to the astronaut. One of the most important, but fortunately, one of the most controllable is the hazard of radiation. There are five main sources of radiation the OTV will have to be protected against: cosmic rays, trapped radiation from the Starfish outer space nuclear explosion, the Van Allen belts, solar flares, and radiation from on-board power (propulsion) systems.

The last two mentioned source should be protected against by shielding surrounding the reactor. In the case of a minor solar flare occurring during flight, the rear of the ship would be pointed toward the sun to provide extra protection given by the thick nuclear reactor shield. In the case of a large solar flare during flight, the ship would also be pointed toward the sun, the mission abandoned, and the ship brought back to the space station as soon as feasible. The other three sources have to be protected against by external shielding surrounding the craft.

The problem of radiation shielding is magnified because nuclear radiation is composed of different components with extremely varying energies - over approximately eleven orders of magnitude. The most important (i.e., most damaging) of these are protons, electrons, and gamma rays. The higher the energy level of any given particle and the greater its flux rate, the higher its possible damage.

Of the first four mentioned sources of nuclear radiation, each is vastly different in terms intensity, duration, and/or make-up. Before a discussion of each, an overview of how

radiation is measured and how much a human can tolerate is
required.

Measuring Radiation

The basic unit of radiation is the roentgen. Although it is
no longer commonly used, its definition is the amount of
radiation necessary to produce 0.001293 grams of air ions
carrying one electrostatic unit of electricity of either sign.

The common term today is the rad (radiation absorbed dose).
It is the radiation of any type corresponding to the absorption
of 100 ergs per gram of any medium. Since the absorbing material
is not stipulated in the definition and every material absorbs
energy at different rates, whatever type of material it is needs
to be specified when the dosage is given.

To complicate matters, each type of radiation constituent
has a varying ability to cause biological damage. Therefore each
constituent is given a factor called the RBE (Relative Biological
Effectiveness) >= 1 which when multiplied by the dose in rads
gives the rem (relative effectiveness, man) dose. This rem dose
is helpful to know since it is independent of the radiation
source type and therefore serves as the means for comparing
radiations and what to protect against.

Some examples of RBEs are as follows:

```
X-Rays, Gamma Rays, and Beta particles        1
Protons > 100 MeV                             1-2
Protons 1.0 MeV < E < 100 MeV                 8.5
Protons 0.1 MeV < E < 1.0 MeV                  10
```

176

```
    Alpha Particles > 5 MeV                          15
    Alpha Particles 1 MeV < E < 5 MeV                20
```

Radiation Effects on Man

The damage done by radiation is directly proportional to the received rem dose. There are variations depending on where on the body the majority of the radiation is received. Critical organs such as the eyes and the liver are much more sensitive to radiation than the body as a whole, however, since they are buried within the body which provides an extra layer of protection and the radiation received on the OTV will impinge on the crew from all directions, the discussion will be limited to whole-body dose and what human whole-body dose tolerance is.

The short-term effects of radiation are about as follows:

Dose in Rads	Probable Effects
10-50	No obvious effects except minor blood changes
50-100	Vomiting and nausea for about 1 day in 5-10% of exposed personnel; fatigue
100-200	Vomiting and nausea for about 1 day followed by other symptoms of radiation sickness in 25-50% of personnel; no deaths

Besides acute problems, radiation also has many long-term consequences. The increase in overall (lung, stomach, etc.) tumor incidence is about 4×10^{-6} per rem per year. The increase in genetic effects is about 0.5×10^{-6} per rem per year, for a total of about 4.5×10^{-6} per rem per year.

As additional guidelines, for the Apollo project for flights up to 30 days a permissible dose of 25 rems was set with a limit of 50 rems resulting in mission termination. The Soviets have similar, but more liberal dose limits. With expected crew rotations, no more than 1 or 2 flights per year per crew member are expected with the OTV. Based on the above considerations, a maximum dose limit of 30-35 rems per mission is planned for as an upper limit on radiation exposure.

Radiation Sources

Galactic, or cosmic radiation provides a continual background source of radiation for the duration of the mission. Its intensity level is low, but is continuous. Cosmic radiation consists mainly of protons with a small fraction of alpha-particles and even smaller fractions of heavier elements.

Cosmic ray energies range from 1×10^{-2} to 1×10^{9} billion electron volts (BeV: note, 1×10^{9} BeV is about 1 joule for a particle weighing 1×10^{-23} kg!) Thankfully flux rate goes down even faster than energy goes up. Fluxes range from one particle per cm^2 per sec. down to 1×10^{-15} particles per cm^2 per sec. With such low intensities at high energy levels, the integrated total expected dosage per seven day mission from cosmic rays is about 5-8 rems.

The Van Allen Belts consist of a second source of radiation. The belts are two torus-shaped regions of ions trapped in the earth's magnetic fields. The centers of these belts are located about 3,000 and 19,000 km from the surface of the earth. The

periphery of the belts are not as distinct and confined and with their low intensity levels present no major threat to the crews. The center of the belts represent a major radiation exposure problem, but with the short transit times through each belt, the total radiation exposure will be within limits.

A mission to geosynchronous orbit and back will involve eight passes through a belt. With the elapsed time in the smaller waiting orbit ranging from 1-3 hours and the trip up to geosynchronous taking approximately 4 hours (yielding maximum belt transit times of 1/2 hour apiece), the maximum total expected dosage per mission from the Van Allen Belts is approximately 10-15 rems.

The electrons trapped from the Starfish multi-megaton outer space nuclear explosion represent a hazard similar to traversing one of the Van Allen Belts. The trapped radiation is located about 3,000 km in altitude and the intensity ranges from 15 rems per day on the edges of the band to 150 rems per day in the center. Each mission will require six passes through the band and with expected velocities, the resulting mission dosage will be about 8-10 rems per mission. The sum of these three sources will expose the crew to a total of 23-33 rems; within the mission allowable dose limits.

The most hazardous source of radiation and unfortunately the most unpredictable comes from solar flares. Solar flares consist mainly of high-energy protons. Particles travelling near the speed of light provide about ten minutes warning of incoming slower-speed protons.

Flares can be divided into about four classes based on intensity. The two most important ones need only be examined since the other two are of such low intensity as not to be a significant radiation hazard. Solar flares are pseudo-random events with probabilities of occurrence. The second worst has only about a 2% probability of occurring during an OTV mission.

The worst-class flare has less than 1% chance of occurring, but if it did it would give the crew a dose of 50 rems or more of radiation if no action on the part of the crew was taken. As mentioned earlier, the best course of action in the event of either class of flare would be to turn the ship's engines towards the sun and use its reactor shielding as extra protection. The reactor's shield would have to be made thicker than otherwise necessary to cope with the radiation from both the reactor and the flare, but since flares are rare, weight is saved by having the extra protection in the reactor shielding rather than in the ship's hull because the reactor shield presents less surface area.

An even better course of action would be not to go up at all if a flare could be predicted in advance. It cannot, but clues exist that hint at increased chances for a flare, such as flares have a greater chance of occurring one solar revolution (the sun revolves on its axis) after the last solar flare. Further study of solar activity may make flare prediction more accurate and reliable with its resulting weight and cost savings.

Shielding Material and Weight

Matter needs to placed between the radiation and the crew to protect them. The more matter between the two, the more protection. Since extra weight means vastly increased costs in terms of fuel, the weight of the shield needs to be minimized. Therefore, for a given level of radiation protection, the material which weighs least is the best choice since manufacturing costs are but a small fraction of the total system cost.

A first-glance choice would be lead for shielding material, but lead has one of the highest weights per given protection. The lowest turns out to be carbon (graphite), but it has the major drawback of debonding under radiation exposure which could lead to structural failure. The next lowest and best candidate is aluminum. The above given rem values per mission are calculated using an aluminum shield 6 grams/cm^2 thick encasing the habitation module, and 4 grams/cm^2 encasing the control module. Mass/area is the standard nomenclature for shielding thickness since for any given area radiation stoppage is proportional to the amount of mass in its path.

The use of 4 and 6 grams/cm^2 thick of aluminum gives a total shield weight of about 19,100 lbs. 19,100 lbs is based on expected OTV surface area and module size. The shield will be similar to a thick aircraft skin in support and construction and will also serve as the pressure hull. It should be noted that a radiation shield of this thickness will also serve as a good meteorite shield. Meteorites travel more than 5 miles per second and their impact can cause great damage (especially with

graphite which "shatters" upon impact, another reason not to use carbon as a shield). The larger the meteorite size the rarer the chance of impact. The particles that need to be protected against are dust-sized ones. Larger ones are much rarer and even if they happened to strike the OTV nothing could be done to reasonably prevent damage with any thickness shield; the OTV would be lost in any case. As an illustration of the relative harmlessness of meteorites because of their rarity, satellites have remained in geosynchronous orbit for many years with less protection than the OTV will get and have continued to function.

STRUCTURE

Dry Weight

Habitation Module Interior
  (bulkheads, galley, etc.) .... 3,000 lbs
Command Module Interior
  (panels, chairs, etc.) .... 800 lbs

Power Systems and ECLSS .... 4,000 lbs

Reaction Control System .... 1,041 lbs

Avionics and Rendezvous Equipment .... 1,039 lbs

Satellite Servicing
  (propellant and hardware) .... 7,900 lbs

Nuclear Reactor and Engine .... 4,000 lbs

Reactor Shielding .... 8,500 lbs

Propellant Tank Structure .... 6,600 lbs

Radiation Shielding/Skin .... <u>19,895 lbs</u>

TOTAL   56,775 lbs

Wet Weight (No Payload)

Propellant Weight without Payload .... 93,292 lbs

TOTAL   150,068 lbs

Weight Weight (15,000 lb Payload)

Propellant Weight with Payload .... 121,184 lbs

Payload .... 15,000 lbs

TOTAL   192,959 lbs

References

1. Bate, R.R., _et al_, <u>Fundamentals of Astrodynamics</u>, Dover Publications, 1971, p 153.

2. <u>Bioastronautics Data Book</u>, pp 424-521.

3. Hagaman, J.A., "A Survey of Structural Material Issues for a Space Station," NASA, 1985, pp 2-19.

4. Pilkey, W.D., _et al_, <u>Mechanics of Solids</u>, Robert E. Krieger Publishing Company, Malabar, Florida, 1986, pp 151-173.

5. Private Communication from Emilio Alfaro-Bou, NASA Langley Research Center, Hampton, Virginia, to J.A. Hagaman.

6. <u>Radiation Shielding in Outer Space</u>, pp 20-end.

7. Rust, David M., "The Great Solar Flares of August, 1972," <u>Sky and Telescope</u>, October, 1972, pp 226-230.

8. Silberberg, R., _et al_, "Radiation Hazards in Space," <u>Aerospace America</u>, vol.25, no.10, October, 1987, pp 38-41.

9. "White-light Solar Flares," <u>Sky & Telescope</u>, January, 1985, p 10.

XI. COST ANALYSIS

Vehicle Cost Analysis

A cost analysis was done for the OTV based on the development costs for the Space Shuttle. The total cost of the Space Shuttle was 13.4 billion dollars (all amounts are in 1984 dollars). This cost included design, development, test and evaluation, and the first manned flight. The dry weight of the Orbiter is approximately 165,000 lbs. This calculates to a cost of $81,210/lb. The weight of the OTV is approximately 50,000 pounds and when multiplied by the cost per pound of the Space Shuttle results in a cost of 4.06 billion dollars. This cost does not include the cost of the nuclear engines which are discussed below.

Propulsion Cost Analysis

There are three types of costs associated with the nuclear propulsion system used onboard the MOVERS OTV.

First, there are the development costs of the engine. The 16,000 pound engine which will be used on the OTV is based upon the actual NERVA design--however, significant development work is required before this engine can be deployed in space. The following figures on the costs associated with this development were obtained from "Nuclear Engine Definition Study--Preliminary Report, Vol. III," published by Los Alamos Laboratories. Those costs were:

186

OVERALL PROGRAM COSTS
($ in thousands)

```
Engineering . . . . . . . . . . . . 298,625
Material and Services . . . . . . . 173,648
Propellant and Pressurants  . . . .  34,810

     TOTAL  . . . . . . . . . . 507,083
```

An additional 21 million dollars was allocated for the modification of existing facilities (i.e., test buildings, tank farms, control buildings, reactor pad, waste effluent clean-up system, etc.). In actuality, significantly more funds will probably have to be allocated to support facilities, as it is currently unknown as to whether these buildings will even exist in 1995. Tentatively, it will be assumed that they will not, and it as been estimated that the cost of building them will be equal to the cost of actually designing and building the small NERVA-derivative engine. Thus:

```
Support Facilities  . . . . . . . . 527,420

    NEW TOTAL COSTS  . . . . . . 1,034,503
```

It must be emphasized that this last figure is totally suspect. It must also be emphasized that these figures are in 1984 dollars.

The second cost associated with the propulsion system is an operational cost associated with the construction and replacement of engines for the OTV. This figure is also difficult to estimate. Los Alamos Laboratories estimated that 103 million dollars would have to be allocated over the course of their nine year development program to purchase the actual hardware for

their various prototypes. As such, this figure does not specify how much the parts for a single engine would cost. Also important to note is that this figure does not include the labor to actually assemble the parts together. With these limitations, only an order of magnitude estimate of the cost of the engine can be provided. It will be assumed that a single engine with a lifetime of at least 10 operating hours will cost 74 million dollars.

Cost of Replacement Engines 73,839

The third cost associated with the propulsion system is the cost of producing and delivering propellant to Low Earth Orbit (LEO). It will tentatively be assumed that the cost of producing propellant is negligible when compared to the cost of delivering it to LEO. As such, estimates of the propellant cost will be based on launch costs alone. Currently it costs $2000 to deliver a pound of anything (i.e., also propellant) to LEO. However, it is questionable as to whether the Space Shuttle would be used for propellant deliveries. In fact, a big dumb booster would probably have to be used; and delivery costs for such a booster have been estimated to be approximately $350/pound.

The cost of shipping this propellant to LEO from the surface of the Earth can be estimated using a range of launch costs. The results are outlined in the following table:

Propellant Requirements & Cost for MOVERS OTV
Hauling 15,000 Pound Payload Roundtrip for LEO to GEO

Wt Propellant		Cost		
Required (lb)	$350/lb	$750/lb	$1000/lb	$2000/lb

(Values are listed in millions)

121,184	42.4	90.9	121.2	242.4

Assuming a fleet of 2 spacecraft, each of which has a minimum design life of 100 missions to GEO--the cost of supplying propellant to these craft was then calculated using the same launch costs:

Wt Propellant		Cost		
Required (lb)	$350/lb	$750/lb	$1000/lb	$2000/lb

(Values are listed in billions)

121,184	8.48	18.2	24.2	48.5

Conclusion

Finally, the estimated overall cost of the OTV will be the sum of the vehicle cost analysis and the propulsion cost analysis. The vehicle cost is 4.06 billion dollars and the propulsion is 1.03 billion dollars. This results in a total cost estimate of 5.09 billion dollars.